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CALCULATED ENGINE PERFORMANCE AND AIRPLANE RANGE FOR VARIETY OF TURBINE-PROPELLER ENGINES

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SUMMARY

An analysis was made of the performance of basic, reheat, regenerative, and regenerative-plus-reheat turbine-propeller engines. The analysis covered an over-all range of flight speeds from 200 to 500 miles per hour, altitudes from 0 to 50,000 feet, and turbine-inlet temperatures from 2000° to 3000° R for a range of compressor pressure ratios from 6 to 42. The effects of flight and engine conditions on fuel consumption, power per square foot of engine frontal area, power per pound of engine weight, and airplane range are shown and compared for several types of turbine-propeller engine.

The analysis indicated that a turbine-propeller engine with regeneration, operating with a regenerative effectiveness of 0.5, gave about 3 percent greater range than did the basic turbine-propeller engine. Greater improvements in range over that indicated by the basic turbine-propeller engine appear possible by use of the more complex engines, such as the reheat and the regenerative-plus-reheat turbine-propeller engines. The turbine-propeller engine with a 100-percent reheat between turbines and with a work distribution between turbines that gave approximately maximum range produced at best (based on calculations that did not include the additional weights required for reheat equipment and controls) about 10-percent improvement in airplane range at low flight speeds and about 15 to 20 percent greater range at a flight speed of 500 miles per hour than the basic turbine-propeller engine. The improvement in range for the regenerative-plus-reheat engine over that of the basic engine was a few percent higher than that given for the reheat engine. For the basic turbine-propeller engine at a flight speed of 500 miles per hour and at an altitude of 30,000 feet, increasing the turbine-inlet temperature from 2000° to 2250° R increased the maximum range about 19 percent; whereas an increase from 2000° to 2500° R indicated a 32 percent increase in range.

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INTRODUCTION

An analysis was made at the NACA Lewis laboratory to determine whether application of reheat or regeneration (or a combination of the two) to the turbine-propeller engine provides appreciable improvement in aircraft performance. The results of this analytical study of engine performance and airplane range for the simple turbine-propeller engine and for variations from the simple, or basic engine (namely, reheat, regenerative, and regenerative-plus-reheat engines) are presented.

The analysis covered a range of flight speeds from 200 to 500 miles per hour and altitudes from 0 to 50,000 feet. The turbine-inlet temperature was varied for most cases from 2000° to 3000° R and, in general, the compressor pressure ratio covered a range sufficiently wide to include the value for maximum airplane range.

SYMBOLS

The following symbols are used throughout this report:

C_v	nozzle velocity coefficient
D_n	nacelle drag, pounds
F	net thrust, pounds
f	fuel-air ratio
hp_c	compressor power, horsepower
hp_t	turbine power, horsepower
J	mechanical equivalent of heat, 778 foot-pounds per Btu
K	ratio of average to initial fuel flow
KR	indicated range (K times R), miles
k	correction factor taking account of change of gas properties due to combustion, pounds air per pound gas
L/D	lift-drag ratio of airplane without nacelles

R airplane range, miles

r_c compressor pressure ratio

1337 $r_{t,1}$ turbine pressure ratio across first turbine

$r_{t,o}$ over-all turbine pressure ratio for cases where reheat is employed

V free-air-stream speed relative to engine, miles per hour

V_o free-air-stream speed relative to engine, feet per second

W_a air flow, pounds per second

W_{ac} accessory weight, pounds

W_{c+t} weight of compressor and turbine, pounds

W_d total disposable load, pounds

W_e power-plant weight including propeller, pounds

W_e' power-plant weight without propeller, pounds

W_f fuel flow, pounds per hour

W_f' initial fuel flow, pounds per mile

W_g gross weight of airplane, pounds

W_{gear} propeller reduction-gear and gearbox weight, pounds

Δh_i ideal enthalpy drop across turbine, Btu per pound air

η_{gear} propeller reduction gear efficiency

η_p propeller efficiency

η_t turbine adiabatic efficiency (based on total pressures)

ENGINES

Basic turbine-propeller engine. - The simplest of the four turbine-propeller engines investigated is the basic engine, which consists of inlet diffuser, compressor, combustor, turbine, and exhaust nozzle. A diagrammatic sketch of this engine is shown in figure 1(a). 1337

Reheat turbine-propeller engine. - The reheat turbine-propeller engine is diagrammatically shown in figure 1(b). The operating cycle of this engine is essentially the same as for the basic engine, except that the expansion occurs in two turbines with additional fuel being burned between the turbines to increase the total turbine work.

Regenerative turbine-propeller engine. - A diagrammatic sketch of the regenerative turbine-propeller engine is shown in figure 1(c). This engine deviates from the basic engine by having the exhaust gases from the turbine directed through a regenerator that heats the air leaving the compressor previous to the addition of fuel in the combustion chamber, which reduces the amount of fuel required to attain the desired turbine-inlet temperature.

Regenerative-plus-reheat turbine-propeller engine. - The regenerative-plus-reheat engine, diagrammatically shown in figure 1(d), combines the features of the previously mentioned engines.

ANALYSIS AND ASSUMPTIONS

Performance of Turbine-Propeller Engines

The assumptions made in establishing the component efficiencies, engine weights, and engine frontal areas for the basic engine operating at current temperature levels are reasonable and attainable in the near future. For the basic engine at higher temperatures and for more complex cycles, however, the assumptions are probably somewhat optimistic.

The following methods and assumptions were used throughout this analysis:

Air flow. - The air flow is assumed to be 5 pounds per second per square foot of engine frontal area for sea-level zero-ram conditions at the compressor inlet. At other altitudes and flight conditions, the air-handling capacity is computed by setting the axial Mach number at the compressor inlet equal to the value corresponding to the sea-level zero-ram condition.

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Inlet diffuser. - The performance of the inlet diffuser was assumed to be the same for all four turbine-propeller configurations. Conditions at the diffuser outlet were determined from the diffuser-inlet conditions by assuming a constant value of ram recovery coefficient (ratio of actual to theoretical pressure rise) of 0.9.

Compressor. - For all engine configurations, the compressor work was calculated from the diffuser-outlet conditions, the compressor pressure ratio, an assumed value of compressor adiabatic efficiency of 0.85, and the data of reference 1. Several conditions were also calculated using a constant small-stage compressor efficiency of 0.88 instead of the constant compressor adiabatic efficiency in order to show the small effect of this assumption on over-all engine performance. These two efficiencies result in the same compressor work at a pressure ratio of 6.

Combustor. - The fuel flow required in the engine combustor to obtain the desired turbine-inlet temperatures was determined by the use of the charts of reference 2. The lower heating value of the fuel was assumed to be 18,900 Btu per pound and the combustion efficiency was assumed to be 0.95 for all cases.

For all conditions where reheat was employed, 100-percent reheat was assumed; that is, the temperature at the inlet to the second turbine was the same as that at the inlet to the first turbine. In order to determine the fuel-air ratio required in the reheat combustor, the charts of reference 2, which are for air, were used directly; the primary combustion process was repeated assigning the same combustion efficiency to the reheat burner as to the primary burner.

Momentum and friction pressure losses in the primary combustor were evaluated in every case and were found to be negligible; consequently, to avoid unnecessary complication and to show maximum performance, no pressure losses were assumed in the reheat combustor.

Turbine. - The turbine horsepower per pound of air per second passing through the turbine was calculated by the relation

$$hp_t/W_a = (1 + f)k\eta_t\Delta h_i J/550 \tag{1}$$

where Δh_i was obtained from reference 1, and the factor k, which takes into account the change in physical properties of the gas during combustion, was obtained from unpublished data. The value of k varied from about 1.00 to 1.02 for the range of conditions investigated. The adiabatic turbine efficiency η_t was taken as 0.9 for the engines without reheat; for engines using reheat, the first and second turbines were each assigned an efficiency of 0.9. For the reheat engines, the gas properties were revaluated and the increased fuel flow was considered.

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In order to determine the proper division of over-all expansion ratio, and hence of total work, between the two turbines of the reheat cycle, a preliminary analysis was made in which the expansion ratio across the first turbine $r_{t,1}$ was arbitrarily related to the over-all turbine expansion ratio $r_{t,o}$ by the relation

$$r_{t,1} = (r_{t,o})^x \qquad (2)$$

where x was assigned values of 0.1, 0.2, 0.3, 0.5, and 0.7. It was found that 0.2 was very close to the optimum value of x, with regard to ultimate airplane range, and therefore $x = 0.2$ was generally used in the present analysis for engines utilizing reheat. Two additional turbine-work divisions are, however, compared with the $x = 0.2$ distribution for several operating conditions. These additional divisions are (1) the expansion ratio of the first turbine is taken as the 0.5 power of the over-all expansion ratio (sometimes this division is referred to as "maximum power reheat") and (2) the expansion ratio across the first turbine is just sufficient to provide the compressor work.

Regenerator. - For the systems using a regenerator, the assumption was made that both the heating and cooling effectiveness of the regenerator were 0.5. An unpublished analysis showed that a regenerator having an effectiveness of 0.5 could be designed with very moderate pressure losses. The assumption was therefore made in the present analysis that no pressure losses occurred in either passage of the regenerator; this assumption results in only slightly optimistic values of engine performance.

Regenerator weights were considered and assumptions regarding these weights are given in the section "Engine Weights."

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Exhaust nozzle. - The gases leaving the turbine (or regenerator) are expanded to atmospheric pressure in a nozzle. The following equation is used to determine the jet velocity for maximum thrust for all four engine configurations:

$$(V_j)_{max} = \frac{C_v^{\ 2}}{\eta_p \eta_{gear} \eta_t} V_0 \qquad (3)$$

This equation is obtained by a simplified analysis of the basic engine for constant turbine, propeller-reduction-gear, and propeller efficiencies and is similar to that derived in reference 3; the equation gives jet velocities in good agreement with those obtained by more precise analysis of both the basic and the more complex engines.

For all the conditions investigated, the exhaust-nozzle velocity coefficient C_v, the turbine adiabatic efficiency η_t, and the propeller-reduction-gear efficiency η_{gear} were assumed to have constant values of 0.97, 0.90, and 0.95, respectively.

Propeller. - The propeller efficiencies used were taken from the appendix of reference 4 and are given in the following table:

Flight Mach number	Propeller efficiency
0.2	0.85
.4	.85
.6	.85
.7	.82
.8	.70

Engine Weights

Turbine-propeller engines. - A survey of the available literature was made in order to determine a representative value for the weight of a turbine-propeller engine to be used as a reference weight. As a result of the survey, the following reference engine specifications were assumed as the basis for the weight calculations used herein; the reference engine is assumed to develop 5000 horsepower at zero-ram sea-level inlet-air conditions, to have a compressor pressure ratio of 6, a frontal area of 8.9 square feet, and a weight of 2500 pounds without the propeller.

An engine weight breakdown indicated that the weight of the reduction gears was about 20 percent of the total engine weight (or 0.1 lb/hp), the combined weight of the turbine and compressor was about 40 percent of the total engine weight, and the remaining 40 percent of the total engine weight consisted of accessories, combustor, nozzle, diffuser, and so forth. 1337

Based on this weight breakdown of the reference engine, the following weight assumptions were used in the analysis:

The reduction gear weight was assumed to vary as the transmitted horsepower according to the relation

$$W_{gear} = 0.1\,\eta_{gear}\,(hp_t - hp_c) \tag{4}$$

The combined weight of the compressor and the turbine was assumed to vary with compressor pressure ratio according to the relation

$$W_{c+t} = 1000\,\frac{\log_e r_c}{\log_e 6} \tag{5}$$

where the factor 1000 represents 40 percent of the reference-engine weight.

The weight W_{ac} of accessories, combustor, nozzle, and so forth is assumed to have a constant value of 1000 pounds, or

$$W_{ac} = 1000$$

The total engine weight is then

$$W_e{}' = 0.1\,\eta_{gear}\,(hp_t - hp_c) + 1000\,\frac{\log_e r_c}{\log_e 6} + 1000 \tag{6}$$

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Equation (6) was used to evaluate the engine weight at each flight condition for all turbine-propeller-engine types. For those engines that required a regenerator, the regenerator weight was taken as 18 pounds per pound of charge air per second at sea level and as 35 pounds per pound of charge air per second at an altitude of 30,000 feet. An unpublished analysis indicated that regenerators with an effectiveness of 0.5 (as used in this analysis) could be designed to meet the foregoing weight specifications. No additional weight was considered for those engines requiring reheat equipment, nor was any weight adjustment made for the cases where turbine-inlet temperature was varied.

The propeller weight was calculated as in reference 4 as a function of the power absorbed, the altitude, and the flight speed for all turbine-propeller configurations.

Airplane Load-Range Characteristics

Airplane ultimate range was calculated as in reference 4. The initial gross weight W_g of the airplane was considered as

$$W_g = (F-D_n)L/D \qquad (7)$$

where the maximum lift-drag ratio L/D of the airplane without nacelles was assumed to be 18. This maximum value of 18 was used as the lift-drag ratio for all flight conditions at which the wing loading did not exceed 80 pounds per square foot. For flight conditions where a lift-drag ratio of 18 resulted in wing loadings greater than 80 pounds per square foot, lift-drag ratio was so adjusted that the wing loading remained constant at 80 pounds per square foot. Engine thrust F and nacelle drag D_n were computed by use of the propeller efficiencies and drag coefficients, respectively, of the appendix of reference 4. Both drag coefficient and propeller efficiency were functions of flight Mach number.

The structure weight of the airplane was assumed as 40 percent of the initial gross weight, and the fuel tank weight as 10 percent of the initial fuel weight. The total disposable load of the airplane (weight of fuel, fuel tank, and cargo) divided by the gross weight of the airplane may then be expressed as

$$\frac{W_d}{W_g} = 0.6 - \frac{W_e}{F-D_n}\frac{D}{L} \tag{8}$$

where W_e is the weight of the engine and propeller.

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The initial fuel flow per pound of gross weight is given by

$$\frac{W_f'}{W_g} = \frac{W_f}{V}\frac{1}{F-D_n}\frac{D}{L} \tag{9}$$

where the fuel flow W_f in pounds per hour is the product of net power and net specific fuel consumption.

When the entire disposable weight is considered as fuel and fuel-tank weight, the maximum indicated range KR is obtained by dividing equation (8) by equation (9), or

$$KR = \frac{\frac{W_d}{W_g}}{1.1\frac{W_f'}{W_g}} \tag{10}$$

The factor 1.1, as previously indicated, accounts for fuel-tank weight. The correction factor K is the ratio of the average to the initial fuel flow and accounts for deviations in flight plan and for progressive reduction in gross weight, and hence, reduction in required fuel flow during the flight. Values of K used herein are based on the Breguet range equation and, as shown in reference 4, are a function only of the ratio W_d/W_g.

RESULTS AND DISCUSSION

The performance of the basic, the reheat, the regenerative, and the regenerative-plus-reheat turbine-propeller engines and the airplane-range characteristics of these turbine-propeller engines are discussed in the following sections:

Performance of Basic Turbine-Propeller Engine

<u>Specific fuel consumption</u>. - The net thrust horsepower specific fuel consumption of the basic turbine-propeller engine plotted against compressor pressure ratio for a turbine-inlet temperature of 2000° R is shown in figure 2(a). Curves are shown for flight speeds of 200, 300, 400, and 500 miles per hour at altitudes of 0, 15,000, 30,000, and 50,000 feet.

As the compressor pressure ratio is increased, the specific fuel consumption decreases until a minimum value is reached after which further increase in pressure ratio results in increased specific fuel consumption. The compressor pressure ratio for minimum specific fuel consumption increases with increasing altitude, being about 15 at sea level and beyond the range investigated at the higher altitudes. The curves are, however, relatively flat and the fuel consumption is within a few percent of the minimum value at a compressor pressure ratio of 20. Specific fuel consumption decreases as altitude is increased over the range investigated.

The specific fuel consumption generally decreases as the flight speed is increased; however, the effect of flight speed is small at all altitudes.

<u>Power per unit engine frontal area</u>. - Corresponding values of net thrust horsepower per square foot of engine frontal area for the basic engine are shown in figure 2(b). As the compressor pressure ratio is increased from the lowest value shown, the power generally increases until a maximum value is reached after which further increase in pressure ratio results in decreased power. The compressor pressure ratio for maximum power increases with increasing altitude, and for a given altitude and flight speed, the maximum power occurs at lower pressure ratios than minimum specific fuel consumption. The power decreases as altitude is increased over the range investigated, principally because of the decreased air-handling capacity of the engine.

The power increases with increasing flight speed; the increase is more pronounced at low than at high compressor pressure ratios. This increase is due mainly to the increase in air flow per unit frontal area as flight speed is increased. At high compressor pressure ratios, however, an increase in flight speed (with resulting

increase in compressor-inlet temperature) causes a greater relative increase in compressor work per pound of air with respect to turbine work per pound of air than at low pressure ratios. This effect, 1337
acting in opposition to the effect of increased air flow, results in a smaller net gain in power for a given increase in flight speed as pressure ratio is increased.

Power per unit engine weight. - The net thrust horsepower per pound of engine weight plotted against compressor pressure ratio is shown in figure 2(c) for the same range of conditions as in figures 2(a) and 2(b). In general, the trends of power per unit engine weight with respect to altitude and flight speed are similar to those of net power per square foot of frontal area shown in figure 2(b). For constant flight velocity, curves of power per unit engine weight decrease with increasing compressor pressure ratio; the decrease is less pronounced as altitude is increased.

Effect of turbine-inlet temperature and compressor efficiency. - Two effects are shown in figures 2(d) to 2(f): (1) the effect of turbine-inlet temperature on engine performance and (2) the effect of using either a constant adiabatic or a constant small-stage compressor efficiency on engine performance.

The effect of turbine-inlet temperature on specific fuel consumption is shown in figure 2(d) where net thrust horsepower specific fuel consumption is plotted against compressor pressure ratio for turbine-inlet temperatures of 2000°, 2250°, and 2500° R at a flight speed of 500 miles per hour and an altitude of 30,000 feet for a constant adiabatic compressor efficiency. Increasing the altitude from 30,000 to 50,000 feet was found to have negligible effect on specific fuel consumption, and consequently the curves for an altitude of 30,000 feet also apply at an altitude of 50,000 feet. Figure 2(d) shows that a decrease of about 12 percent in specific fuel consumption is obtained by increasing the turbine-inlet temperature from 2000° to 2500° R, if the increase in temperature is accompanied by a corresponding increase in pressure ratio.

Also shown in figure 2(d) are curves of specific fuel consumption that were calculated using a small-stage compressor efficiency for the same flight speed and altitudes and for turbine-inlet temperatures of 2000°, 2500°, and 3000° R. The small-stage compressor efficiency was taken as 0.88, which results in the same compressor work at a compressor pressure ratio of 6 as the adiabatic compressor efficiency of 0.85 and hence the same engine performance. This plot indicates about the same percentage decrease in specific fuel consumption as obtained with constant adiabatic efficiency for an increase in turbine-inlet temperature from 2000° to 2500° R.

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Curves of net thrust horsepower per square foot of engine frontal area corresponding to the specific fuel consumptions of figure 2(d) are shown in figure 2(e). With a constant compressor adiabatic efficiency, an increase in the turbine-inlet temperature from 2000° to 2500° R results in an increase in the maximum thrust horsepower of about 60 percent at both 30,000 and 50,000 feet with an increase in corresponding pressure ratio from 8 to 14. Also plotted on figure 2(e) are curves calculated using a small-stage compressor efficiency for the same flight speed and altitude and for turbine-inlet temperatures of 2000°, 2500°, and 3000° R. As with the constant adiabatic efficiency, an increase in maximum thrust horsepower of 60 percent is obtained for an increase in turbine-inlet temperature from 2000° to 2500° R at both altitudes shown.

Corresponding values of net thrust horsepower per pound of engine weight are shown in figure 2(f). In all cases the power per pound of engine weight decreases with increasing compressor pressure ratio. When compared at the compressor pressure ratio corresponding to maximum power per unit frontal area, the power per unit weight is increased 25 percent for an increase in turbine-inlet temperature from 2000° to 2500° R at both altitudes. Curves of net power per pound of engine weight calculated using a small-stage compressor efficiency are included for comparison and again show only small differences from the curves for constant adiabatic efficiency.

Performance of Reheat Turbine-Propeller Engine

The reheat engine, although more complex than the basic engine, offers the possibility of substantially increased power output. In the present analysis, as previously indicated, the pressure ratio of the first turbine was chosen to be equal to the 0.2 power of the over-all turbine expansion ratio; and the reheat was assumed to be 100 percent; that is, enough fuel was added between the turbines to bring the turbine-inlet temperature at the inlet of the second turbine to the same value as that at the inlet to the first turbine.

Specific fuel consumption. - In figure 3(a), net thrust horsepower specific fuel consumption is plotted against compressor pressure ratio for a turbine-inlet temperature of 2000° R. Curves are shown for flight speeds of 200, 300, 400, and 500 miles per hour at altitudes of 0, 15,000, 30,000, and 50,000 feet. The curves, which have trends similar to those for the basic engine, indicate approximately the same specific fuel consumptions as for the basic engine at a compressor pressure ratio of 6 and somewhat lower values than for the basic engine at a pressure ratio of 20.

Power per unit engine frontal area. - Corresponding net thrust horsepower per square foot of engine frontal area for the reheat engine are shown in figure 3(b). Again the curves have about the same shape as those for the basic engine; however, maximum power occurs at higher pressure ratios for the reheat cycle and is roughly 20 percent greater than that of the basic engine at all conditions.

Power per unit engine weight. - The net thrust horsepower per pound of engine weight for the reheat engine is shown in figure 3(c) plotted against compressor pressure ratio for flight speeds of 200, 300, 400, and 500 miles per hour at altitudes of 0, 15,000, 30,000, and 50,000 feet. The curves are similar in shape to those for the basic engine but indicate larger values of power per unit weight at all conditions. As previously stated, however, no increase was made in engine weight to account for turbine staging, additional combustors, or additional structures and consequently the curves are a direct result of the increase in the power of the reheat engine over that of the basic engine.

Effect of turbine-inlet temperature. - In figure 3(d), the net thrust horsepower specific fuel consumption of the reheat engine is plotted against compressor pressure ratio for turbine-inlet temperatures of 2000°, 2250°, and 2500° R at a flight speed of 500 miles per hour and an altitude of 30,000 feet. As for the basic engine, the same curves apply at 50,000 feet. The curves are similar in shape to those for the basic engine, but slightly lower specific fuel consumptions are indicated for the reheat engine at all conditions.

Net thrust horsepower per square foot of engine frontal area for the same range of variables is shown in figure 3(e). Increasing the turbine-inlet temperature from 2000° to 2500° R increases the maximum thrust power about 60 percent at both 30,000 and 50,000 feet as was also found for the basic engine.

Corresponding curves of net thrust horsepower per pound of engine weight are shown in figure 3(f). Increasing the turbine-inlet temperature from 2000° to 2500° R increases the power per unit weight about 30 percent at both 30,000 and 50,000 feet at compressor pressure ratios corresponding to maximum power per unit frontal area.

Effect of turbine power division. - In order to illustrate the effect on performance of various power divisions between the two turbines of the reheat engine, three configurations of the reheat engine are compared in figure 3(g) on the basis of net thrust horsepower specific fuel consumption and net thrust horsepower per square
1337 foot of engine frontal area for a range of compressor pressure ratios. The curves are for a flight speed of 200 miles per hour at sea level and a turbine-inlet temperature of 2000° R; the general shape of the curve is, however, similar at other flight speeds and altitudes. The three reheat engines considered are: (1) an engine in which the pressure ratio of the first stage is equal to the 0.2 power of the over-all available turbine pressure ratio; (2) an engine in which the first-turbine-stage expansion ratio is taken as the 0.5 power of the over-all turbine expansion ratio; (3) an engine in which the expansion ratio of the first turbine stage is so selected as to do only the compressor work. In all three types of engine, a turbine adiabatic efficiency of 0.90 was used for each stage regardless of work distribution. Corresponding curves for the basic engine are also shown in figure 3(g).

The engine in which the first turbine does the required compressor work results in the highest specific fuel consumption, whereas the engine using the 0.2 power distribution gives the lowest specific fuel consumption over the range of compressor pressure ratios considered (fig. 3(g)). The engine with the 0.5 power distribution results in the highest power per unit frontal area, whereas the engine with the 0.2 power distribution results in the lowest power per unit area of the three engines. The engine with the 0.2 power distribution gives about 10 percent lower specific fuel consumption and 13 percent lower horsepower per unit engine frontal area than the engine with 0.5 power distribution.

The engine wherein the compressor is driven by the first turbine stage (usually the turbine and the compressor are considered coupled by a separate shaft with the second turbine delivering power directly to the propeller) is desirable from control and component matching considerations. Both this system and the engine with the 0.5 power distribution are, however, omitted from further considerations inasmuch as a preliminary study showed these engines to be inferior in airplane range performance when compared with the engine with the 0.2 power division.

Comparing the minimum specific fuel consumption shown for the reheat engine using the 0.2 power distribution with the minimum shown for the basic engine indicates about an 8-percent decrease.

The maximum thrust horsepower shown for the reheat engine with the 0.2 power distribution is 22 percent greater than the maximum indicated for the basic engine.

Performance of Regenerative Turbine-Propeller Engine

Specific fuel consumption. - In figure 4(a) net thrust horsepower specific fuel consumption for the regenerative turbine-propeller engine is plotted against compressor pressure ratio for a turbine-inlet temperature of 2000° R at flight speeds of 200, 300, 400, and 500 miles per hour and altitudes of 0 and 30,000 feet. Minimum specific fuel consumption occurs at compressor pressure ratios of approximately 6 at sea level and 10 at 30,000 feet. These pressure ratios for minimum fuel consumption are lower than those for any of the previous engines. The maximum decrease in minimum specific fuel consumption of this engine from that of the basic engine is 4 percent at sea level and less than 1 percent at 30,000 feet. When the specific fuel consumptions of the two engines are compared at a pressure ratio of 8, the improvement of the regenerative engine over that of the basic engine is about 7 and 14 percent at altitudes of 0 and 30,000 feet, respectively, for all flight speeds investigated.

Power per unit engine frontal area. - Net thrust horsepower per square foot of engine frontal area is not presented for the regenerative engine, inasmuch as the only significant change from that of the basic engine is the difference in fuel flow. The power per unit area of the regenerative engine was found to be within 1 percent of the basic engine values (fig. 2(b)).

Power per unit engine weight. - Corresponding net thrust horsepower per pound of engine weight is shown in figure 4(b). Because of the added weight of the regenerator, the power per unit engine weight is lower than that of the basic engine at every operating condition.

Effect of turbine-inlet temperature. - The variation of net thrust horsepower specific fuel consumption with compressor pressure ratio and turbine-inlet temperature at a flight speed of 500 miles per hour and an altitude of 30,000 feet is shown in figure 4(c). The compressor pressure ratio for minimum specific fuel consumption increases from about 8 at 2000° R to 14 at 2500° R with a corresponding reduction of approximately 10 percent in specific fuel consumption.

The power per unit frontal area is not shown because, as previously explained, figure 2(e) for the basic engine gives sufficiently accurate values for the regenerative engine.

1337 The net thrust horsepower per pound of engine weight is shown in figure 4(d) plotted against compressor pressure ratio. A gain of 30 percent is indicated for a change in turbine-inlet temperature from 2000° to 2500° R at pressure ratios corresponding to minimum specific fuel consumption.

Performance of Regenerative-Plus-Reheat Turbine-Propeller Engine

Specific fuel consumption. - The variation of net thrust horsepower specific fuel consumption with compressor pressure ratio for the regenerative-plus-reheat engine is shown in figure 5(a). Curves are shown for flight speeds of 200, 300, 400, and 500 miles per hour at altitudes of 0 and 30,000 feet. At sea level, the combination of regeneration and reheat gives a decrease in minimum specific fuel consumption of approximately 2 percent from that of the reheat engine (fig. 3(a)), 6 percent from that of the regenerative engine (fig. 4(a)), and 10 percent from that of the basic engine (fig. 2(a)). At 30,000 feet the corresponding decreases in fuel consumption are approximately 4, 5, and 7 percent for the reheat, regenerative, and basic engines, respectively. The compressor pressure ratios for minimum specific fuel consumption for the regenerative-plus-reheat engine, which are 8 at sea level and 12 at 30,000 feet, are lower than those for both the reheat and the basic engines and slightly higher than for the regenerative engine at both altitudes.

Power per unit engine frontal area. - The corresponding net thrust horsepower per square foot of engine frontal area for the regenerative-plus-reheat engine is accurately represented by the curves of figure 3(b) for the reheat engine.

Power per unit engine weight. - The net thrust horsepower per pound of engine weight is shown in figure 5(b). The values are slightly lower than those for the reheat engine because of the addition of the weight of the regenerator.

Effect of turbine-inlet temperature. - The variation of net thrust horsepower specific fuel consumption with compressor pressure ratio and turbine-inlet temperature at a flight speed of 500 miles per hour and an altitude of 30,000 feet is shown in figure 5(c). The pressure ratio for minimum specific fuel consumption increases from about 12 at a turbine-inlet temperature of 2000° R to about 20 at a temperature of 2500° R with a corresponding reduction of 8 percent in minimum specific fuel consumption.

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Net thrust horsepower per pound of engine weight is shown in figure 5(d). These curves are the same as those of the reheat engine except for the effect of added regenerator weight.

Airplane-Range Characteristics with Turbine-Propeller Engines

For a given flight speed and altitude, minimum fuel consumption, maximum power per unit frontal area, and maximum power per unit weight may occur at widely different compressor pressure ratios, as shown in figures 2 to 5. These performance parameters are all related to airplane range, so that if ultimate airplane range is used as the criterion of engine performance, the compressor pressure ratio corresponding to maximum ultimate range can be considered to be the value at which the parameters are properly weighted. In the following discussion of airplane-range characteristics, the range for each flight speed and altitude corresponds, except for a few cases as previously indicated, to a compressor pressure ratio defined in the foregoing way.

Basic engine. - The range characteristics of an airplane powered by the basic turbine-propeller engine for a turbine-inlet temperature of 2000° R are shown in figure 6(a), where the dimensionless ratio of total disposable load to gross weight W_d/W_g is plotted against the initial fuel flow per ton of gross weight $2000\ W_f'/W_g$ for a range of flight speeds at altitudes of 0, 15,000, 30,000, and 50,000 feet. A separate symbol is used for each flight speed, and the compressor pressure ratios for ultimate range are tabulated on the figure.

The range is given in this manner because, as shown in reference 4, such plots afford a means of presenting not only the range for various conditions of flight speed, altitude, and pay load but also show the effects of changes in many of the pertinent airplane and engine design variables.

The ultimate range, that is the range for which the entire disposable load consists of fuel and tank weights, is obtained from

the slope of a line drawn from the origin through any operating point. (See fig. 6(a).) The slope of such a line is equal to the ratio of the disposable load to the initial fuel rate and after multiplication by the factor 2000/1.1 becomes the indicated range
1337 KR. (See section "ANALYSIS AND ASSUMPTIONS.") For convenience, scales for estimating indicated range KR and ratio of average to initial fuel flow K are included. In figure 6(a), the maximum ultimate range (for the conditions investigated) occurs at a flight speed of 200 miles per hour and an altitude between 15,000 and 30,000 feet where a KR of about 7100 miles is indicated. The value of K for the corresponding disposable load is 0.73, giving an actual maximum range of about 9800 miles. At constant altitude, the range decreases with increasing flight speed.

The allowable pay load (the pay load is the difference between the total disposable weight and the weight of the required fuel plus fuel tanks) for a specific range less than the ultimate range may also be estimated from figure 6(a) as follows: A line is drawn from the origin to the desired range, for example to KR equals 1000 miles (fig. 6(a)). The vertical distance from a given speed-altitude operating point to this line is the pay load per pound of gross weight, and the vertical distance below this line is the weight of the fuel and tank per pound of gross weight. The value of K corresponding to this value of fuel plus tank weight per unit gross weight is obtained from the plot on the left side of the figure and is used to calculate the actual range.

The ultimate range and the corresponding compressor pressure ratio for each flight-speed - altitude point are tabulated in table I for the basic and succeeding engine types.

The effect of increasing the turbine-inlet temperature on the range characteristics of the basic engine is shown in figure 6(b) for a flight speed of 500 miles per hour at altitudes of 30,000 and 50,000 feet. Curves are shown for temperatures of 2000°, 2250°, and 2500° R with the compressor pressure ratio being varied from 6 to 20, 6 to 32, and 6 to 42, respectively, as indicated on the figure. In every case, increasing pressure ratio over the range investigated causes both disposable load and fuel flow to decrease. At an altitude of 30,000 feet, increasing the turbine-inlet temperature from 2000° to 2250° R increases the ultimate range about 19 percent, whereas an increase from 2000° to 2500° R results in only a 32 percent increase, which indicates that range increases at a decreasing rate as temperature increases. Similarly, at 50,000 feet, increasing the turbine-inlet temperature from 2000° to 2250° R increases the ultimate range about 25 percent, whereas an increase from 2000° to 2500° R results in an increase of 47 percent. The increase in range due to the increase in turbine-inlet temperature is accompanied by an increase in compressor pressure ratio from 14 at a turbine-inlet temperature of 2000° R to 20 at a temperature of 2250° R and to 26 at a temperature of 2500° R for an altitude of 30,000 feet. At an altitude of 50,000 feet, the pressure ratio increases from 8 at a turbine-inlet temperature of 2000° R to 14 at a temperature of 2250° R and to 20 at a temperature of 2500° R. 1337

Also shown in figure 6(b), are curves obtained by use of a constant small-stage compressor efficiency. These curves are approximately the same as those shown for a constant adiabatic efficiency.

No increase in engine weight with increased turbine-inlet temperature was assumed; however, an increase in engine weight would reduce the advantage of using high turbine-inlet temperatures.

Reheat engine. - The range characteristics for the reheat engine are shown in figure 7(a) with the curves for the basic engine superimposed as dashed lines.

For some conditions, the compressor pressure ratio corresponding to ultimate range for the reheat engine with a turbine-inlet temperature of 2000° R was found to be greater than 20. For these cases, the range corresponding to a pressure ratio of 20 is shown. This limit on pressure ratio results in a maximum deviation from the true ultimate range of less than 5 percent. About 10-percent improvement in range resulting from the use of reheat is noted at the low flight speeds and gains of about 15 to 20 percent are found at a flight speed of 500 miles per hour. As previously indicated, the performance calculations for this engine (and also for the regenerative-plus-reheat engine) do not include the additional weights required for reheat equipment and controls and the turbine efficiencies assumed are probably optimistic for this engine.

The effect of increasing the turbine-inlet temperature on the range characteristics of the reheat engine is shown in figure 7(b). At an altitude of 30,000 feet, an increase in range of about 15 percent is achieved by increasing the temperature from 2000° to 2250° R. Increasing the temperature from 2000° to 2500° R causes an increase in range of approximately 27 percent. At an altitude of 50,000 feet, the respective increases in range with temperature are 20 and 40 percent. The ultimate range and corresponding compressor pressure ratio for each flight-speed - altitude point are tabulated in table I.

Regenerative engine. - A comparison of the range characteristics of the regenerative turbine-propeller engine with those of the basic engine is made in figure 8(a). The gain in ultimate range obtained by using regeneration over the maximum range of the basic engine is about 3 percent for all flight conditions.

The effect of increasing turbine-inlet temperature on range characteristics of the regenerative engine is shown in figure 8(b). The improvement in range at a flight speed of 500 miles per hour and an altitude of 30,000 feet obtained by increasing the turbine-inlet temperature from 2000° to 2250° R is 16 percent; an increase in turbine-inlet temperature from 2000° to 2500° R gives a 31-percent gain. Tabulated values of ultimate range and corresponding compressor pressure ratio are given in table I.

Regenerative-plus-reheat engine. - In figure 9(a), the range characteristics of the regenerative-plus-reheat engine are compared with that of the basic engine. The regenerative-plus-reheat engine shows a value of KR of about 8100 miles with a K of 0.73, giving an ultimate range of about 11,100 miles at a flight speed of 200 miles per hour at an altitude of 30,000 feet, as compared with an ultimate range of 9800 miles (table I) indicated for the basic engine at the same conditions.

The improvement in range of the regenerative-plus-reheat engine over that of the basic engine, for all conditions shown, is a few percent higher than the improvement in range of the reheat engine over that of the basic engine (table I).

In figure 9(b), at a flight speed of 500 miles per hour and an altitude of 30,000 feet a gain of about 15 percent in range is obtained when the turbine-inlet temperature is increased from 2000° to 2250° R; an increase in turbine-inlet temperature from 2000° to 2500° R gives a 27 percent gain.

SUMMARY OF RESULTS

The results of an analytical investigation of the comparative performances and range characteristics of a variety of turbine-propeller engines (basic, reheat, regenerative, and regenerative-plus-reheat engines) may be summarized as follows:

1. Comparison of the minimum specific fuel consumption of the regenerative-plus-reheat turbine-propeller engine with the simpler turbine-propeller engines showed that at sea level the combination

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of regeneration and reheat gave a decrease in minimum specific fuel consumption of approximately 2 percent from that of the reheat engine 6 percent from that of the regenerative engine, and 10 percent from that of the basic engine. At an altitude of 30,000 feet, the corresponding decreases in fuel consumption were 4, 5, and 7 percent, respectively. The compressor pressure ratios for minimum specific fuel consumption for the regenerative-plus-reheat engine, which were 8 at sea level and 12 at an altitude of 30,000 feet, were lower than those for both the reheat and basic engines and slightly higher than for the regenerative engine at both altitudes.

2. The range of the turbine-propeller engine increased at a decreasing rate as the turbine-inlet temperature was increased, (no increase in engine weight was assumed to accommodate increases in turbine-inlet temperatures). For the basic turbine-propeller engine at a flight speed of 500 miles per hour and an altitude of 30,000 feet, increasing the turbine-inlet temperature from 2000° to 2250° R increased the maximum range about 19 percent, whereas an increase from 2000° to 2500° R indicated a 32 percent increase in range. These increases in range, however, were accompanied by increases in compressor pressure ratio.

3. A turbine-propeller engine with 100-percent reheat between turbines, and with a work distribution between turbines that gave approximately maximum range, resulted in at best about 10-percent improvement in airplane range at low flight speeds and about 15 to 20 percent greater range at a flight speed of 500 miles per hour than the basic turbine-propeller engine. This trend was constant as altitude was changed. The performance calculations did not include the additional weights required for reheat equipment and controls, and the turbine efficiencies assumed were probably optimistic for this engine when compared with the basic engine.

4. A turbine-propeller engine with regeneration, operating with a regenerative effectiveness of 0.5, gave about 3 percent greater range than did the basic turbine-propeller engine at all flight speeds and altitudes.

5. A turbine-propeller engine operating with a combination of regeneration and reheat indicated a slightly greater improvement in range over that of the basic turbine-propeller engine at all flight speeds and altitudes than that found for the reheat engine.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio, January 10, 1950.

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2. Pinkel, Benjamin, and Karp, Irving M.: A Thermodynamic Study of the Turbojet Engine. NACA Rep. 891, 1947.

3. Kühl, H.: Preliminary Report on the Fundamentals of the Control of Turbine-Propeller Jet Power Plants. NACA TM 1172, 1947.

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TABLE I - ULTIMATE RANGE OF AIRPLANE WITH TURBINE-PROPELLER ENGINE

(a) Turbine-inlet temperature of 2000° R

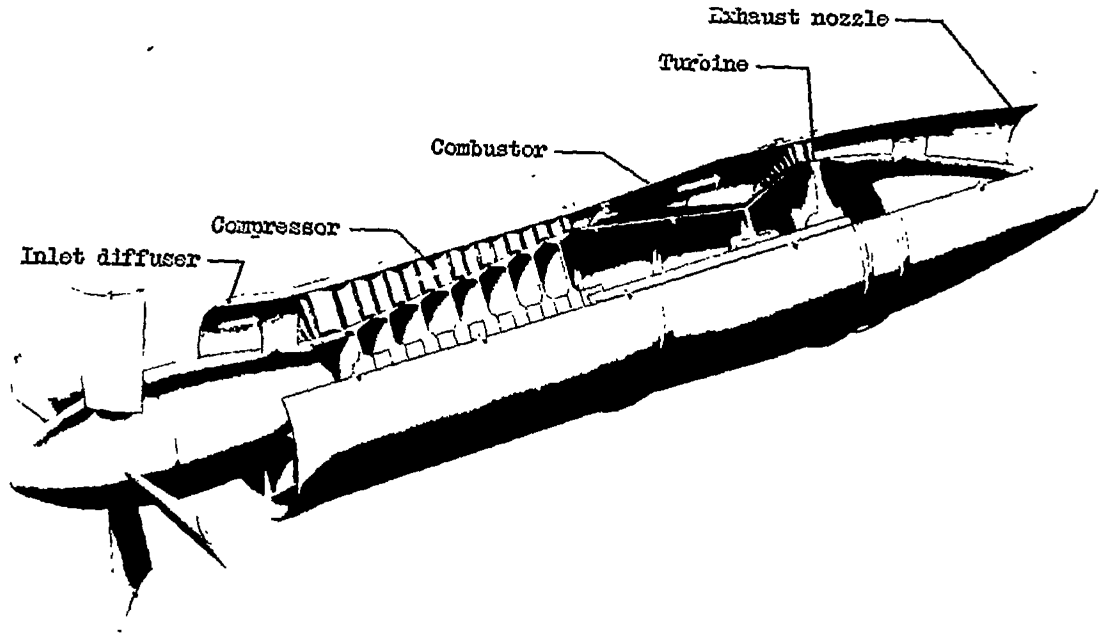
	Flight speed (mph)	Altitude (ft)							
		0		15,000		30,000		50,000	
		Range (miles)	Compressor pressure ratio	Range (miles)	Compressor pressure ratio	Range (miles)	Compressor pressure ratio	Range (miles)	Compressor pressure ratio
Basic turbine-propeller engine	200 300 400 500	9300 7100 4000 2200	12 14 12 8	9800 9200 6400 3700	14 14 14 10	9800 9300 8500 5400	14 14 14 14	7600 6700 6000 4500	14 14 12 8
Reheat turbine-propeller engine[1]	200 300 400 500	10,300 7900 4500 2500	20 20 20 14	10,700 10,100 7100 4200	20 20 20 20	10,700 10,300 9400 6200	20 20 20 20	8500 7600 7000 5400	20 20 20 14
Regenerative turbine-propeller engine	200 300 400 500	9600 7300 4200	8 8 6			10,200 9600 8700 5500	12 10 10 8		
Regenerative-plus-reheat turbine-propeller engine[1]	200 300 400 500	10,400 7900 4600 2500	10 8 8 8			11,100 10,500 9600 6200	14 14 14 12		

[1]First turbine pressure ratio equal to 0.2 power of available pressure ratio.

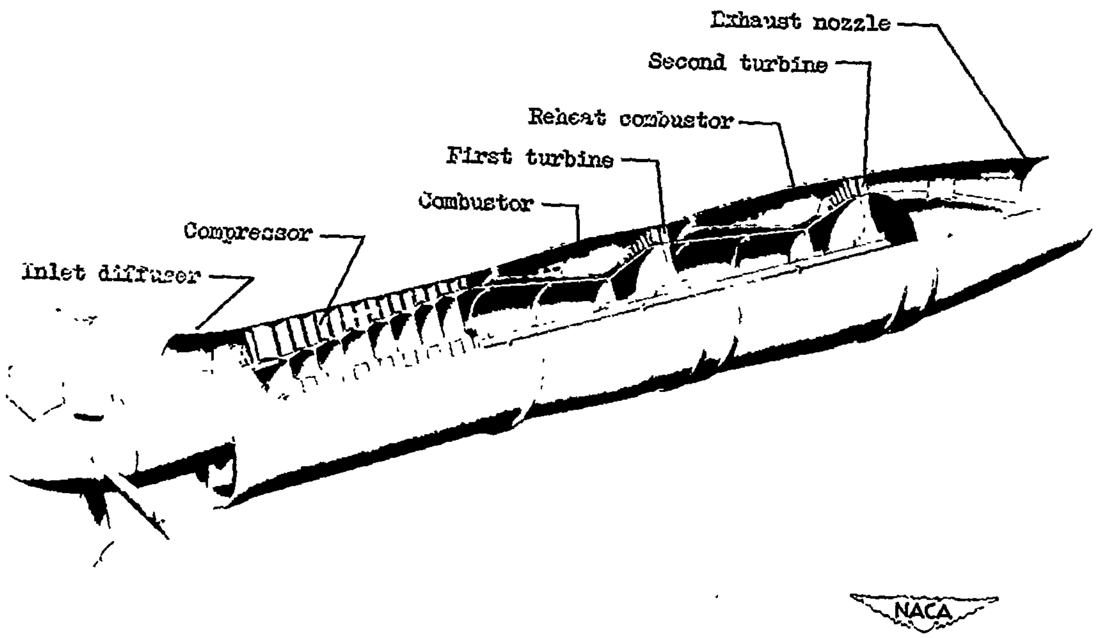
(b) Turbine-inlet temperatures of 2250° and 2500° R for flight speed of 500 miles per hour

	Turbine-inlet temperature (°R)	Altitude (ft)			
		30,000		50,000	
		Range (miles)	Compressor pressure ratio	Range (miles)	Compressor pressure ratio
Basic turbine-propeller engine	2250 2500	6400 7100	20 26	5600 6600	14 20
Reheat turbine-propeller engine[1]	2250 2500	7100 7900	32 42	6500 7600	22 32
Regenerative turbine-propeller engine	2250 2500	6400 7200	10 12		
Regenerative-plus-reheat turbine-propeller engine[1]	2250 2500	7100 7900	14 16		

[1]First turbine pressure ratio equal to 0.2 power of available pressure ratio.

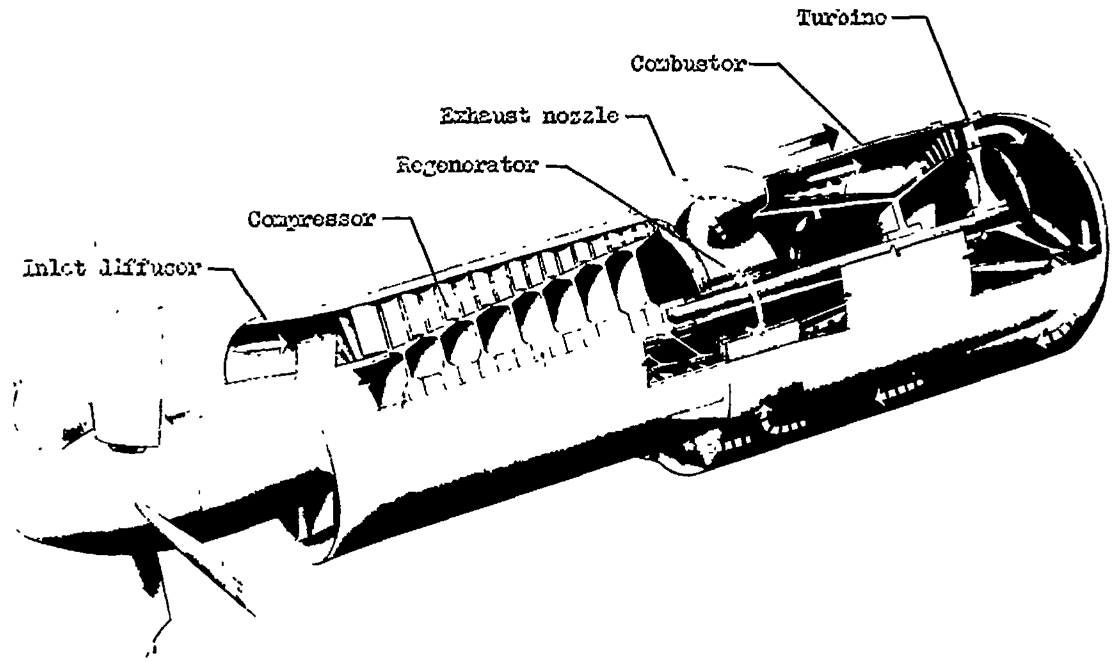


(a) Basic turbine-propeller engine.

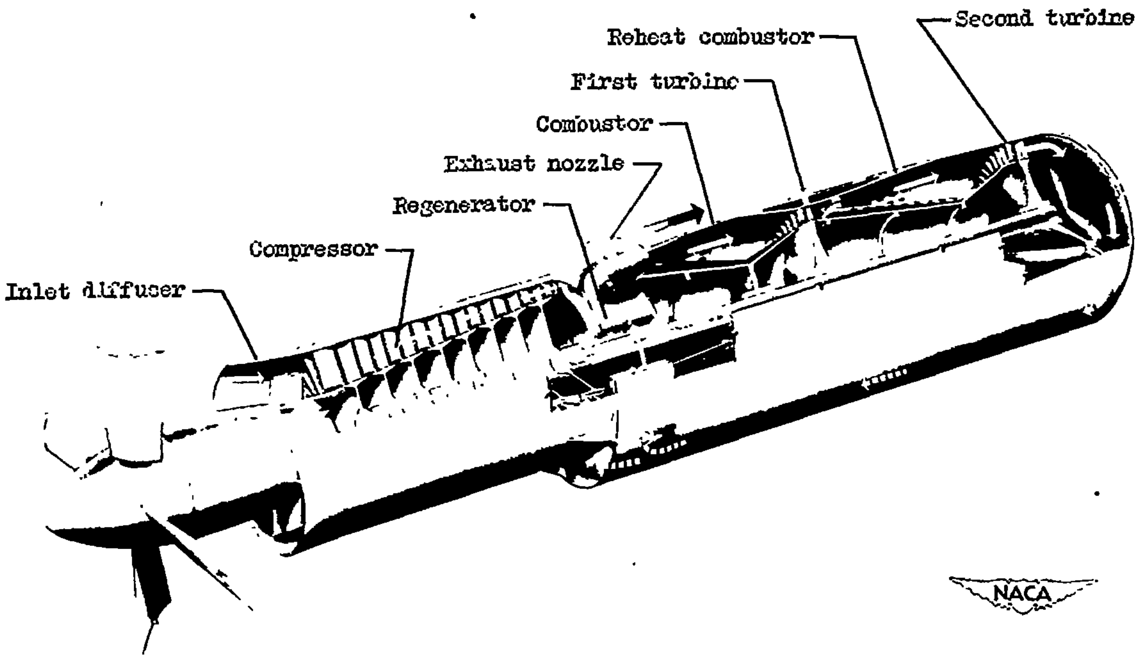


(b) Reheat turbine-propeller engine.

Figure 1. - Diagrammatic sketch of turbine-propeller system.

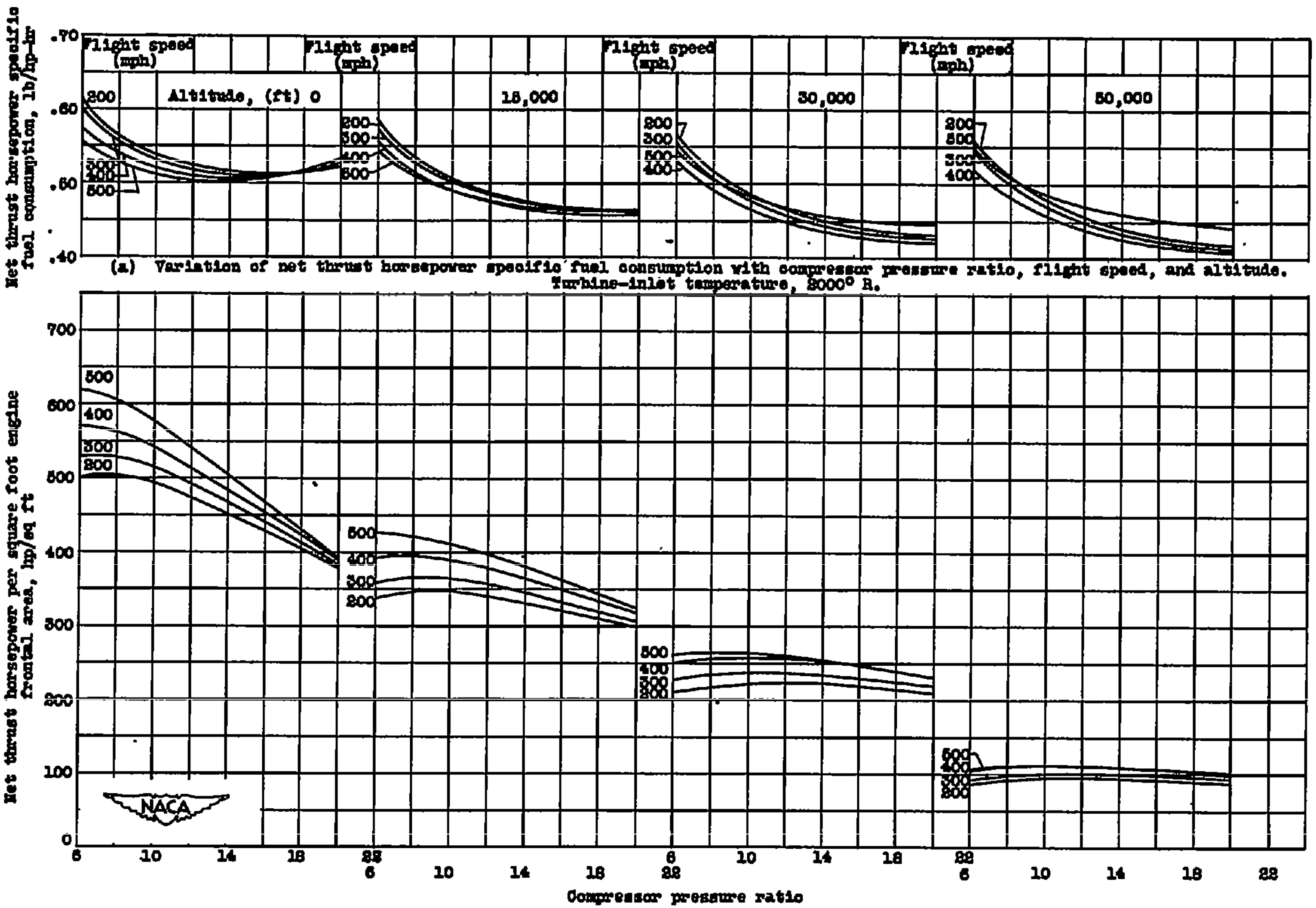


(c) Regenerative turbine-propeller engine.



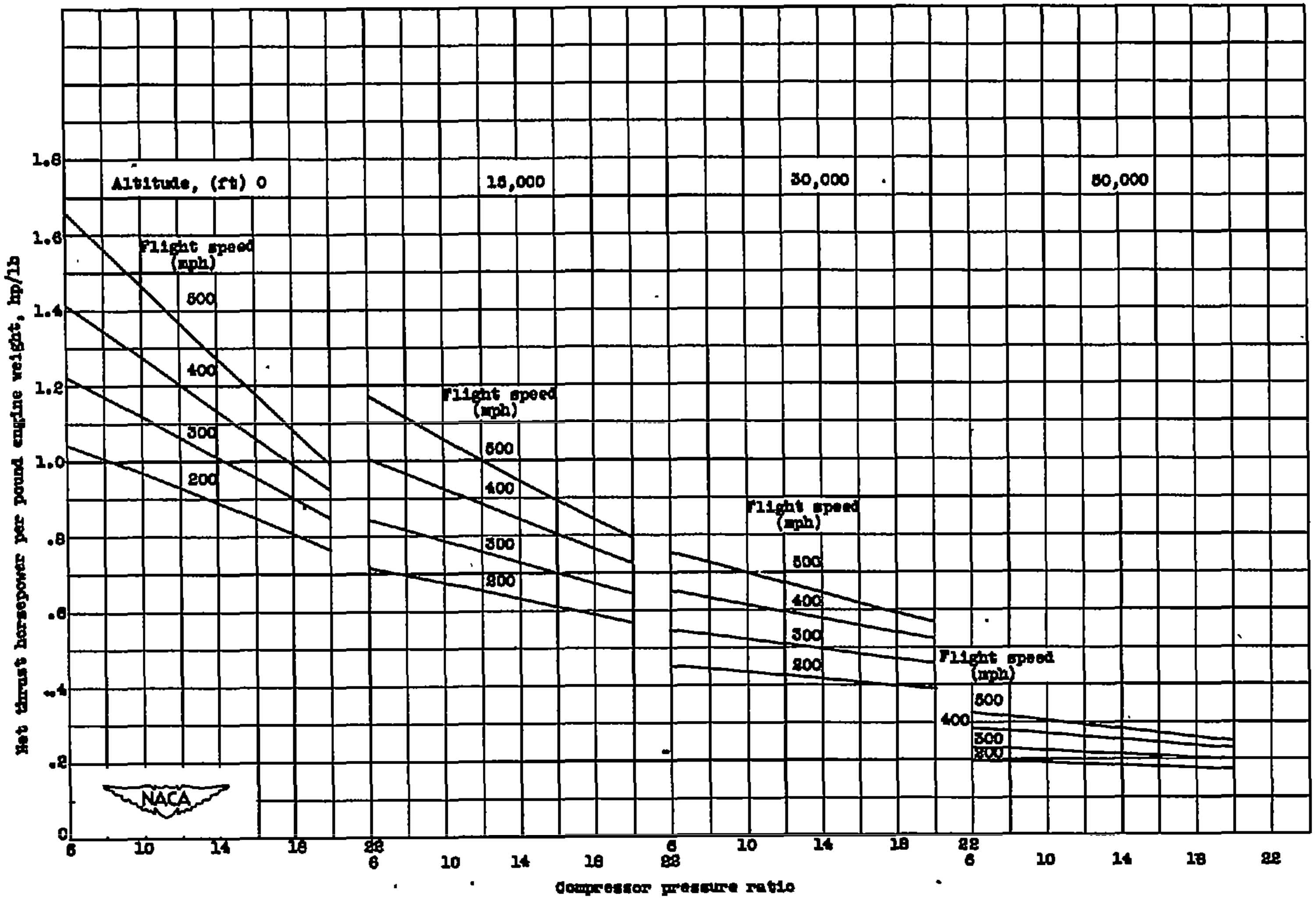
(d) Regenerative-plus-reheat turbine-propeller engine.

Figure 1. - Concluded. Diagrammatic sketch of turbine-propeller system.



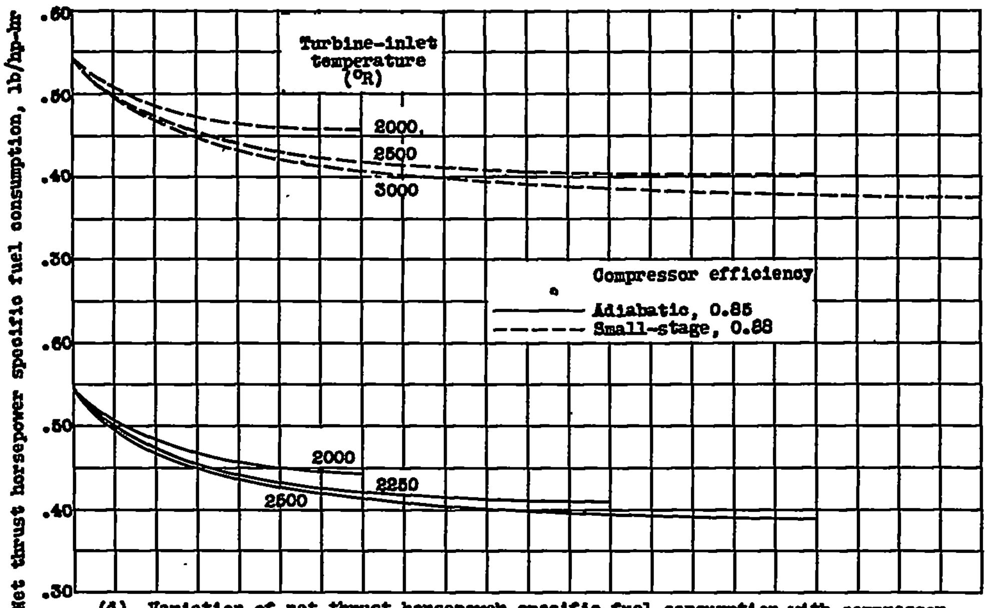
(b) Variation of net thrust horsepower per square foot of engine frontal area with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R.

Figure 8. - Continued. Performance of basic turbine-propeller engine.

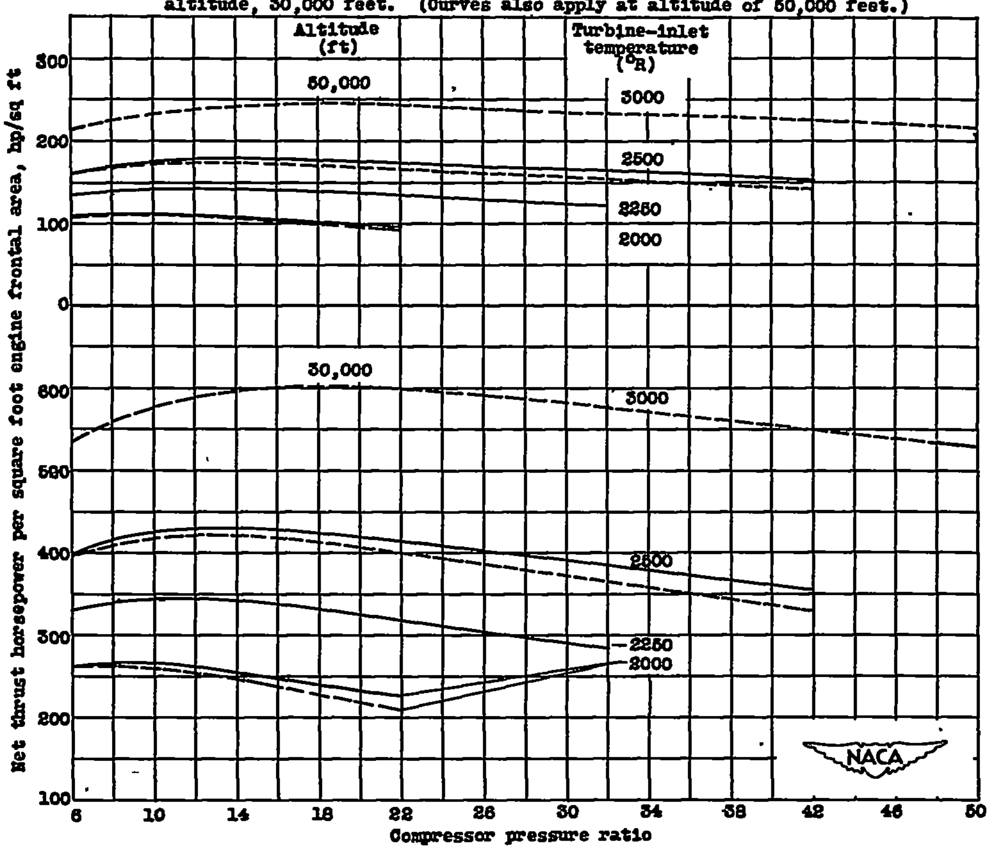


(c) Variation of net thrust horsepower per pound of engine weight with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R.

Figure 8. - Continued. Performance of basic turbine-propeller engine.

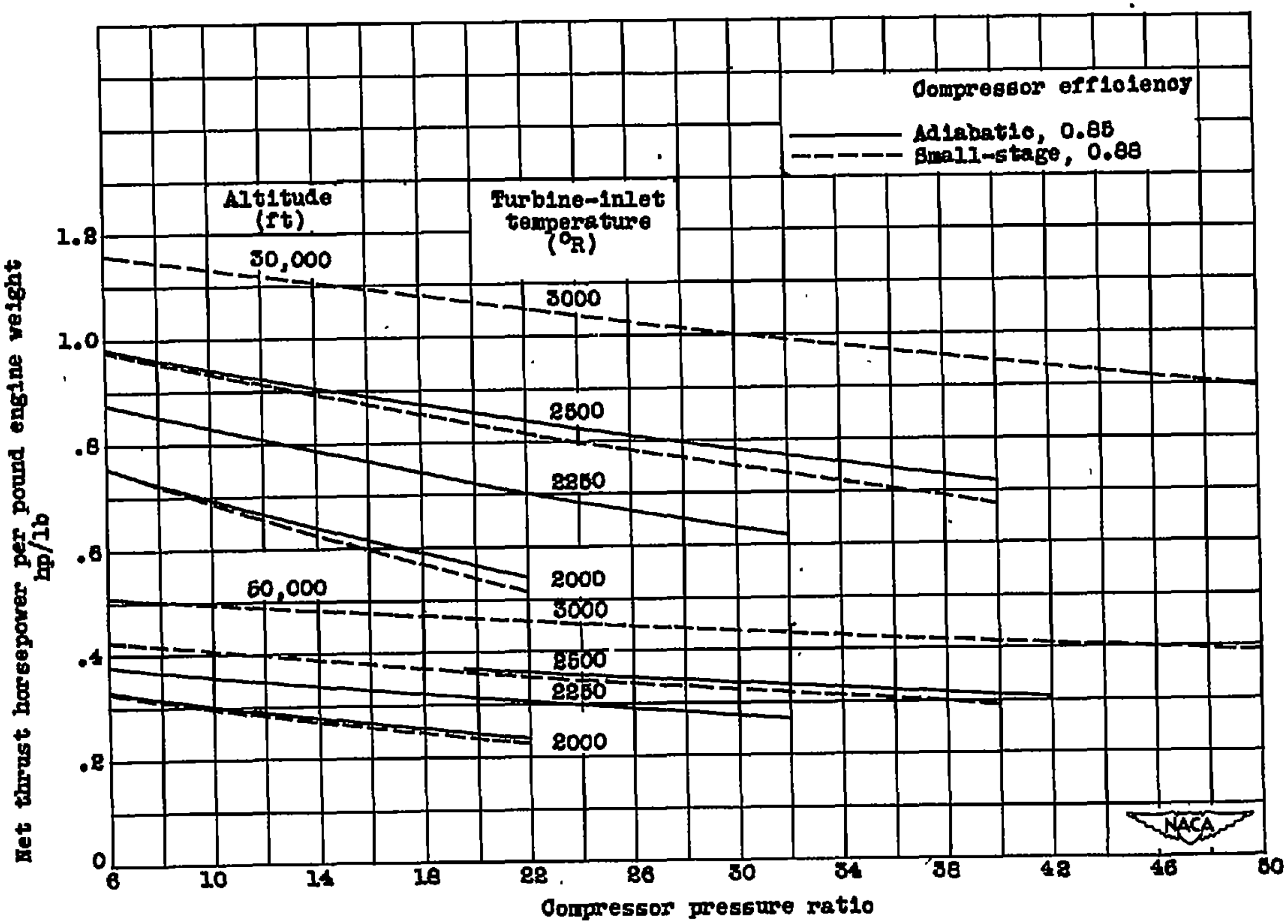


(d) Variation of net thrust horsepower specific fuel consumption with compressor pressure ratio and turbine-inlet temperature for cases of adiabatic and small-stage compressor efficiencies. Flight speed, 500 miles per hour; altitude, 30,000 feet. (Curves also apply at altitude of 50,000 feet.)



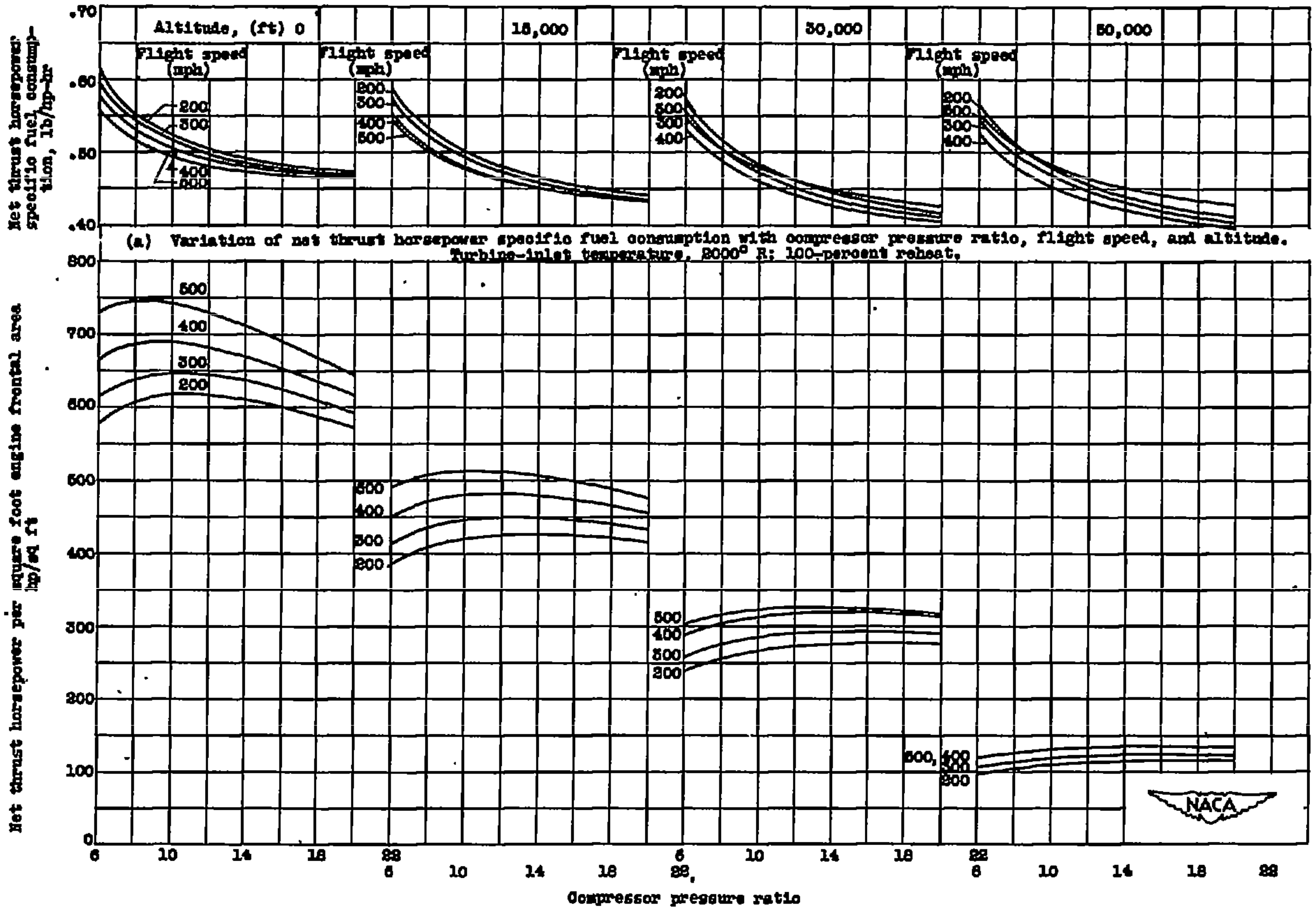
(e) Variation of net thrust horsepower per square foot of engine frontal area with compressor pressure ratio and turbine-inlet temperature for cases of adiabatic and small-stage compressor efficiencies. Flight speed, 500 miles per hour; altitude, 30,000 and 50,000 feet.

Figure 2. - Continued. Performance of basic turbine-propeller engine.



(f) Variation of net thrust horsepower per pound of engine weight with compressor pressure ratio and turbine-inlet temperature for cases of adiabatic and small-stage compressor efficiencies. Flight speed, 500 miles per hour; altitude, 30,000 and 50,000 feet.

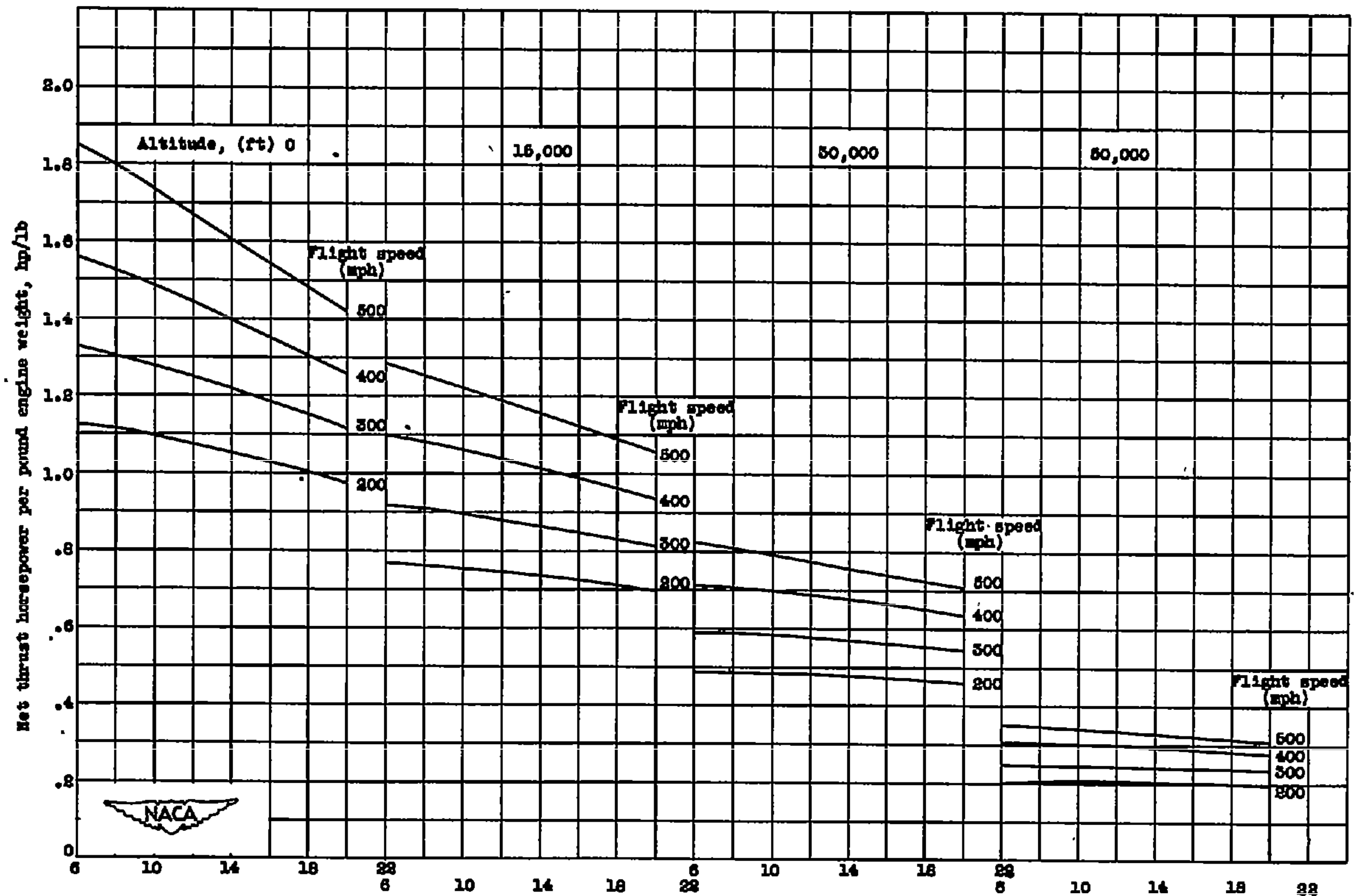
Figure 2. - Concluded. Performance of basic turbine-propeller engine.



(a) Variation of net thrust horsepower specific fuel consumption with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R; 100-percent reheat.

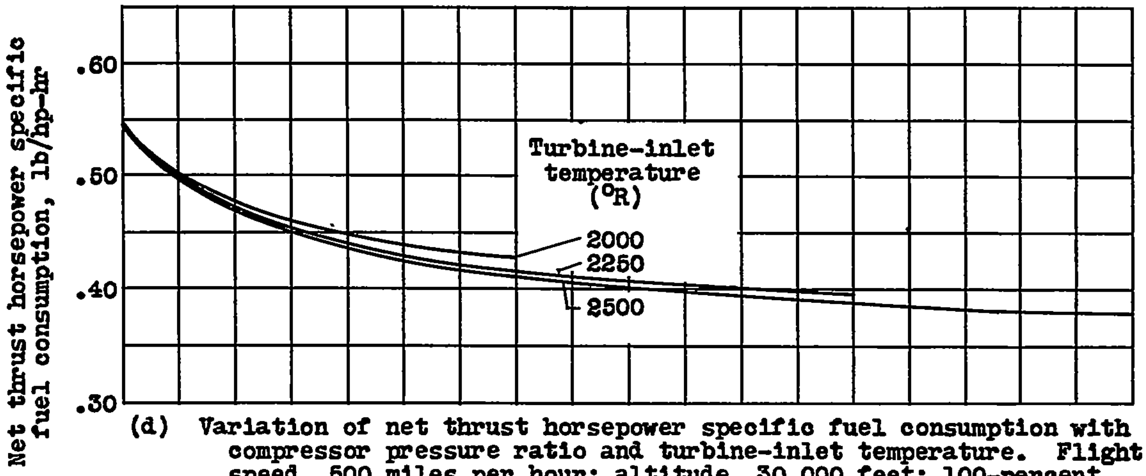
(b) Variation of net thrust horsepower per square foot of engine frontal area with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R; 100-percent reheat.

Figure 3. - Performance of reheat turbine-propeller engine.

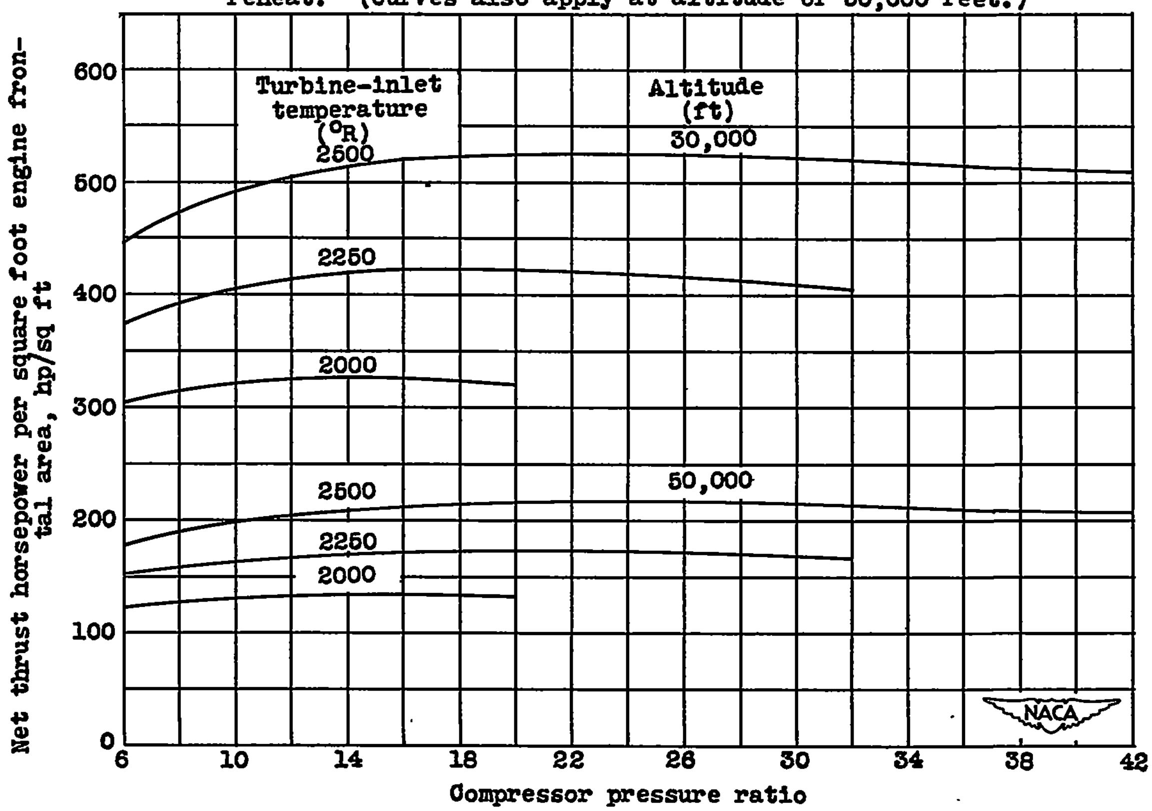


(c) Variation of net thrust horsepower per pound of engine weight with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R; 100-percent reheat.

Figure 3. - Continued. Performance of reheat turbine-propeller engine.

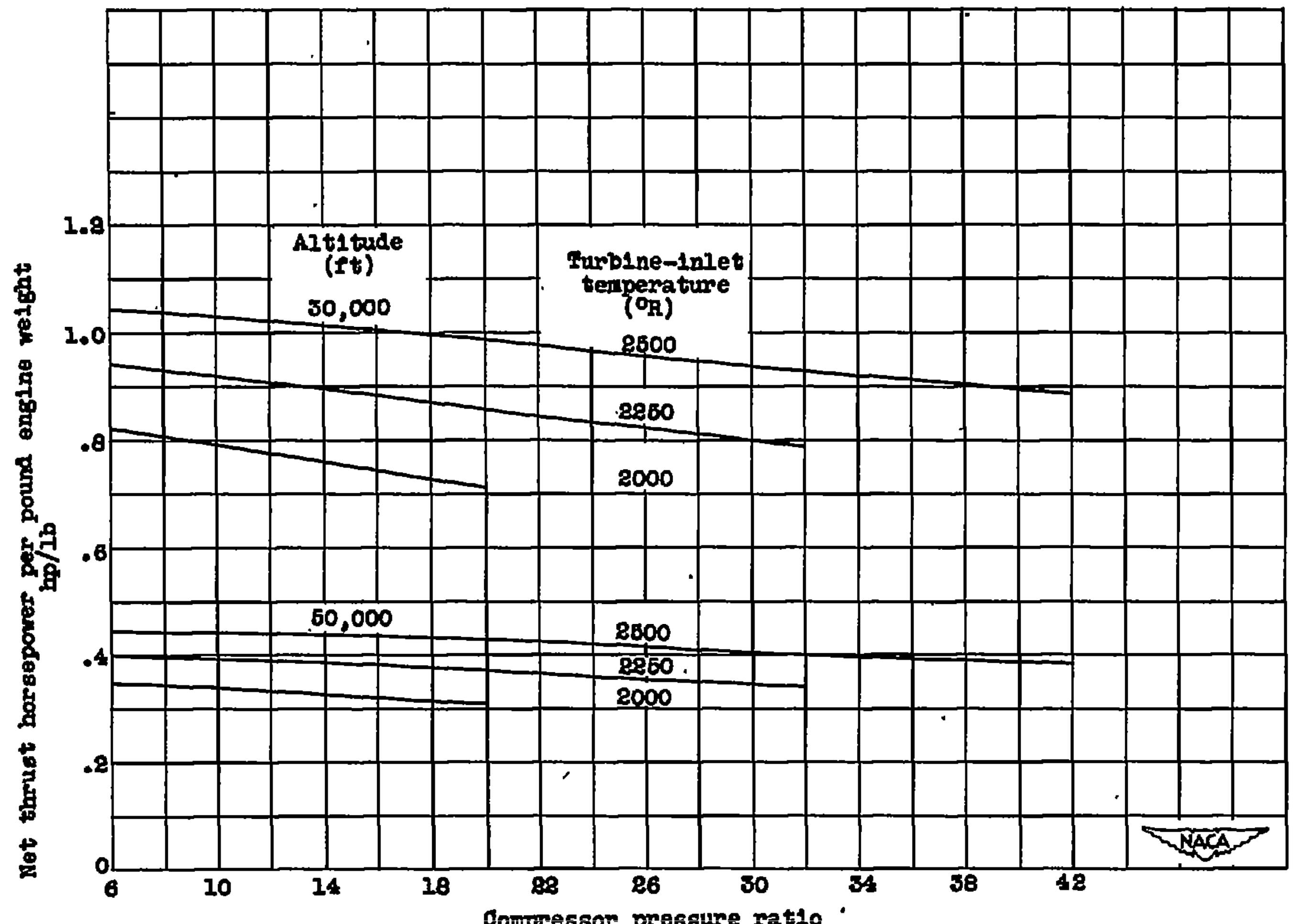


(d) Variation of net thrust horsepower specific fuel consumption with compressor pressure ratio and turbine-inlet temperature. Flight speed, 500 miles per hour; altitude, 30,000 feet; 100-percent reheat. (Curves also apply at altitude of 50,000 feet.)



(e) Variation of net thrust horsepower per square foot of engine frontal area with compressor pressure ratio and turbine-inlet temperature. Flight speed, 500 miles per hour; altitude, 30,000 and 50,000 feet; 100-percent reheat.

Figure 3. - Continued. Performance of reheat turbine-propeller engine.



(f) Variation of net thrust horsepower per pound of engine weight with compressor pressure ratio and turbine-inlet temperature. Flight speed, 500 miles per hour; altitude, 30,000 and 50,000 feet; 100-percent reheat.

Figure 3. - Continued. Performance of reheat turbine-propeller engine.

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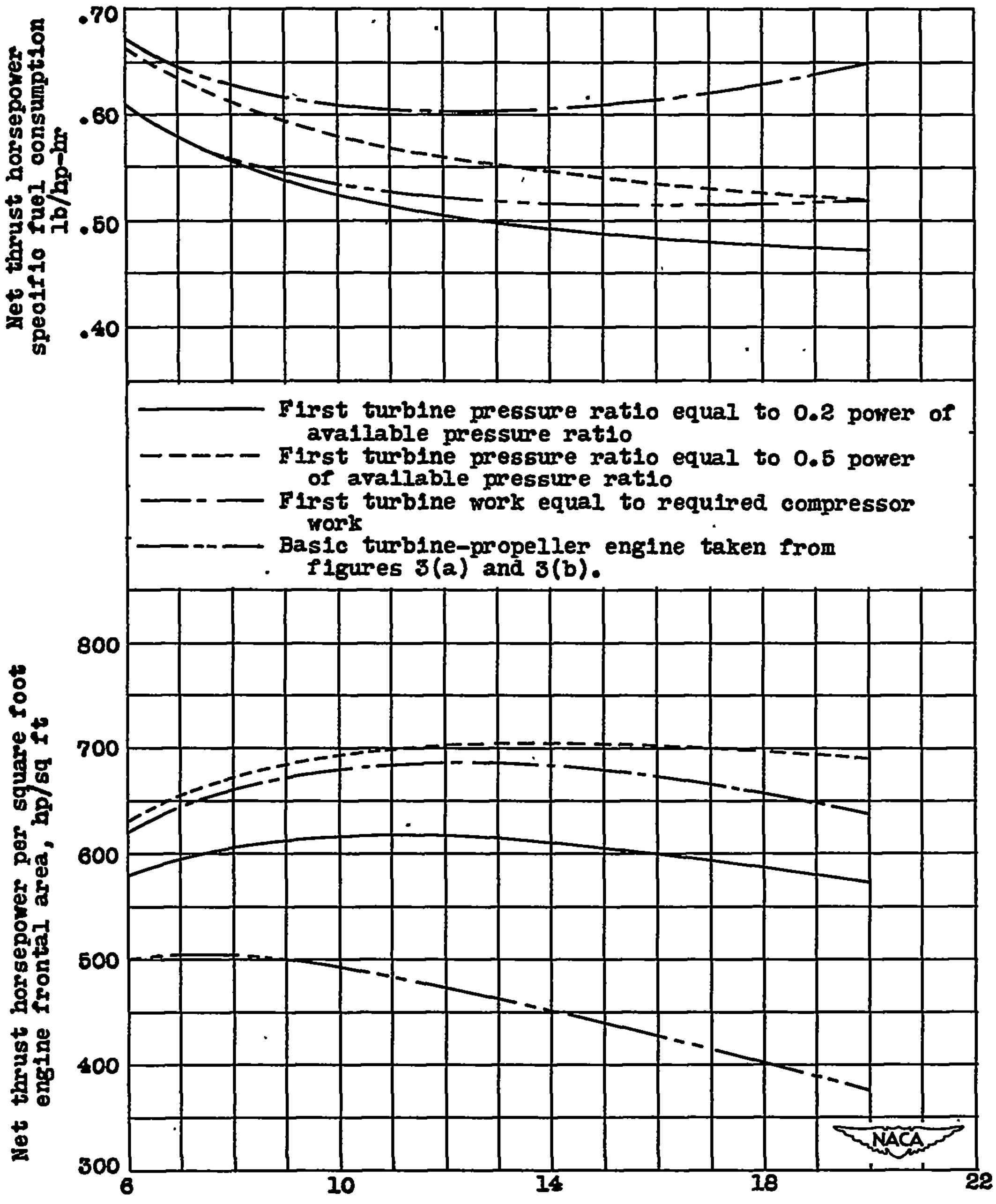
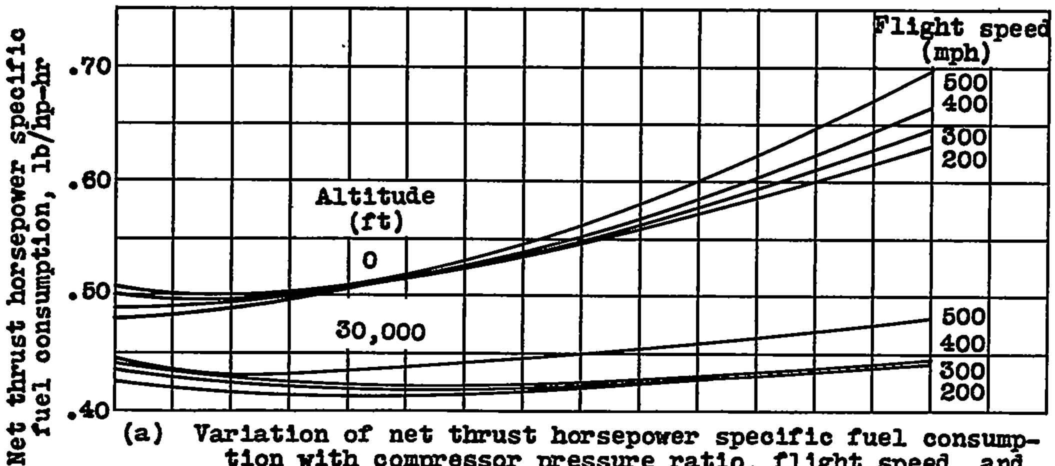
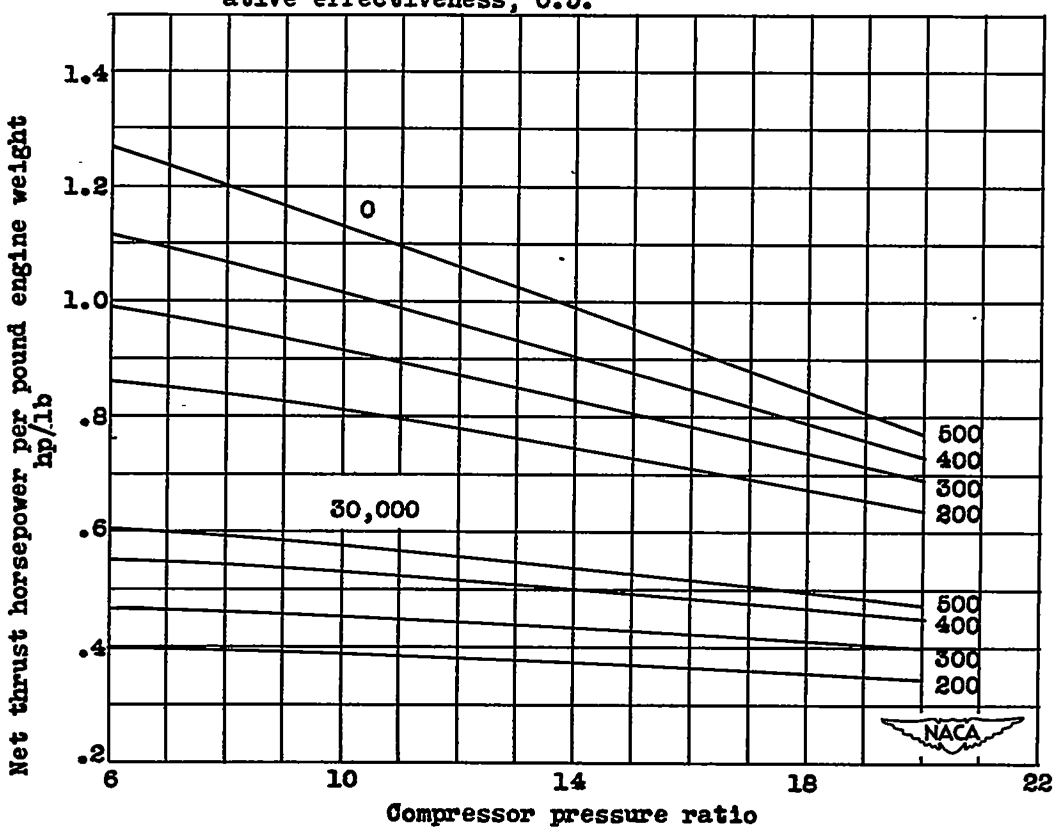


(g) Effect of various turbine power distributions for range of compressor pressure ratios. Flight speed, 200 miles per hour; altitude, 0 feet; turbine-inlet temperature, 2000° R; 100-percent reheat.

Figure 3. - Concluded. Performance of reheat turbine-propeller engine.



(a) Variation of net thrust horsepower specific fuel consumption with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R; regenerative effectiveness, 0.5.



(b) Variation of net thrust horsepower per pound of engine weight with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R; regenerative effectiveness, 0.5.

Figure 4. - Performance of regenerative turbine-propeller engine.

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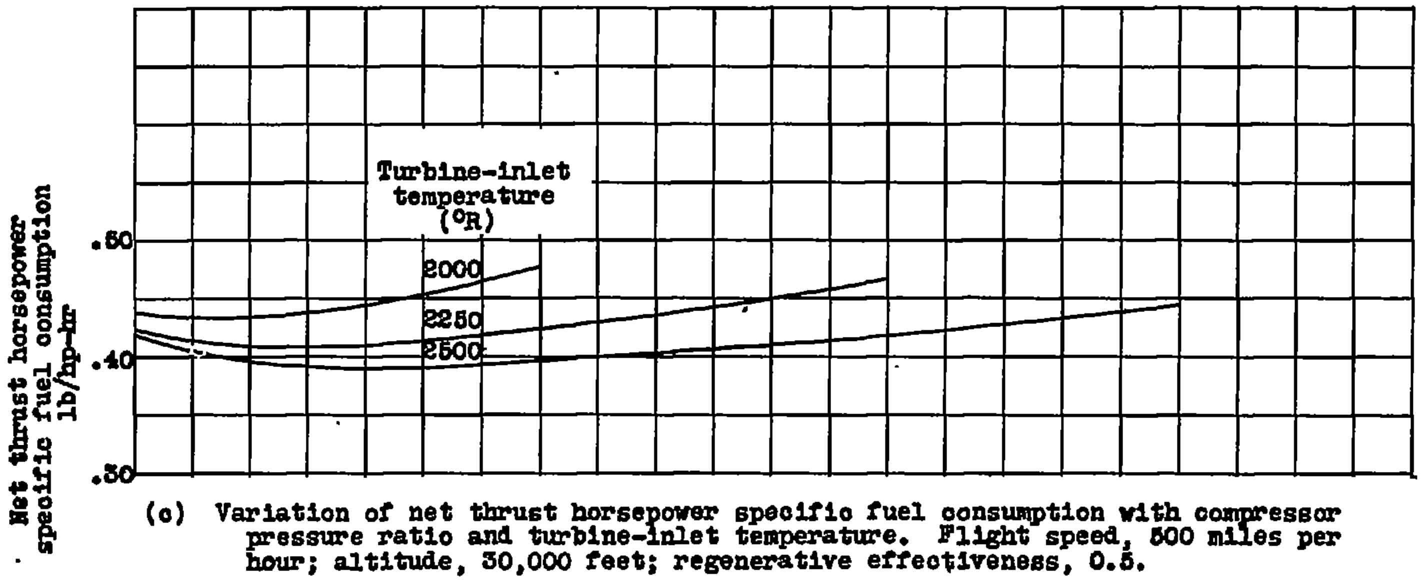
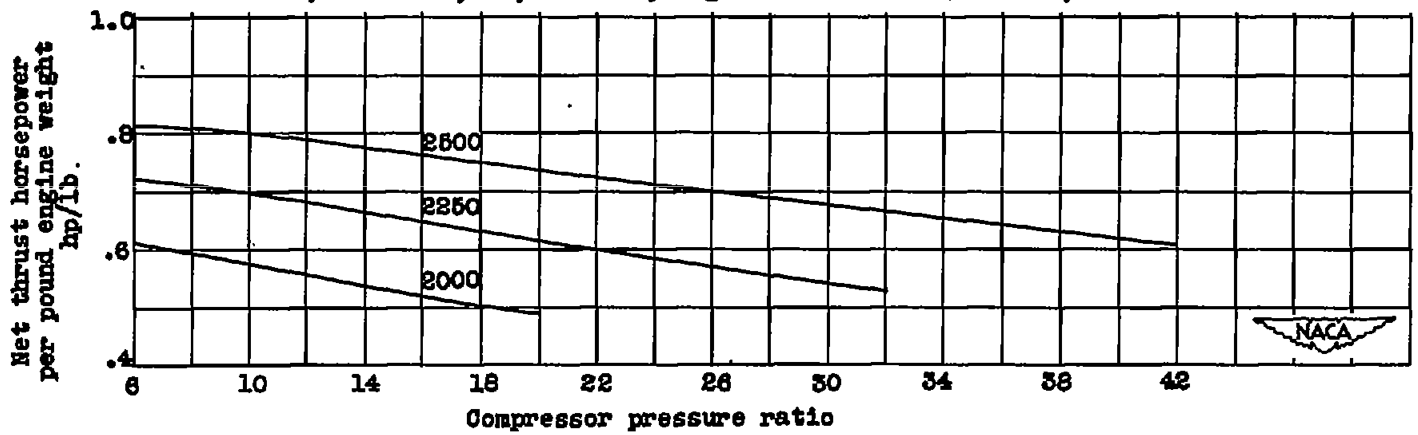
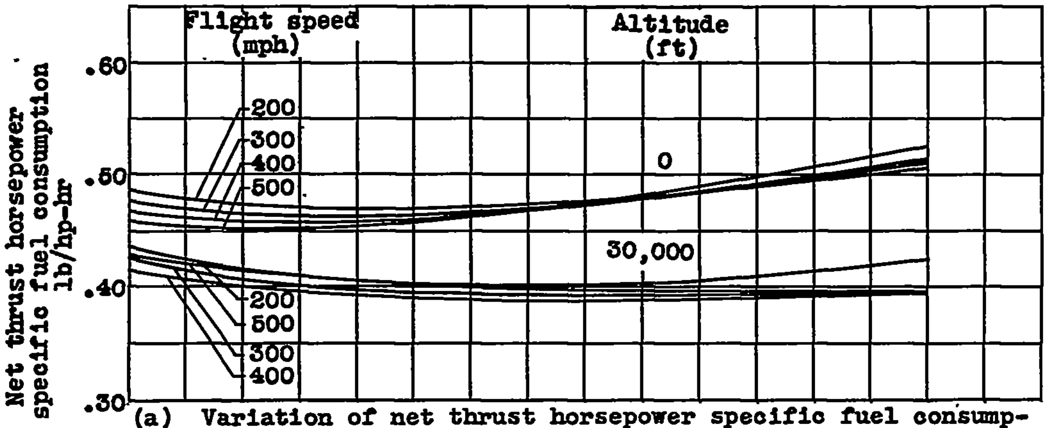


(c) Variation of net thrust horsepower specific fuel consumption with compressor pressure ratio and turbine-inlet temperature. Flight speed, 500 miles per hour; altitude, 30,000 feet; regenerative effectiveness, 0.5.

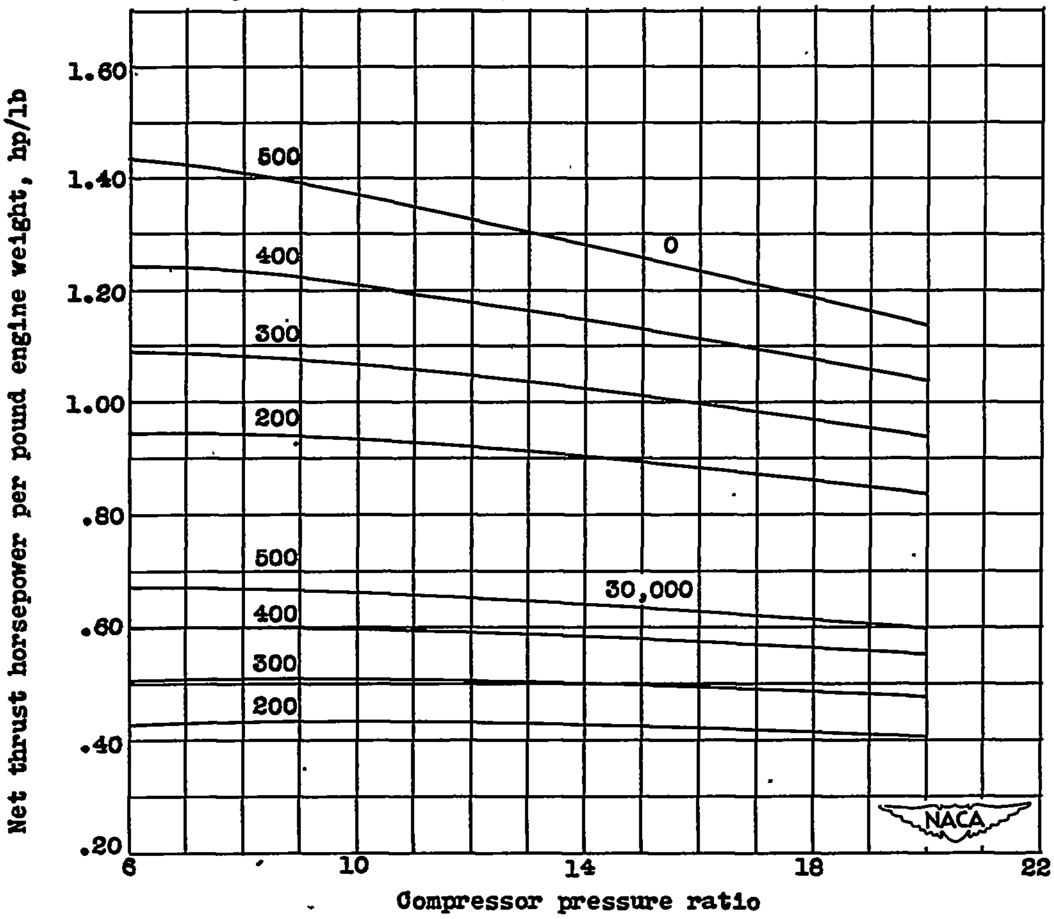


(d) Variation of net thrust horsepower per pound of engine weight with compressor pressure ratio and turbine-inlet temperature. Flight speed, 500 miles per hour; altitude, 30,000 feet; regenerative effectiveness, 0.5.

Figure 4. - Concluded. Performance of regenerative turbine-propeller engine.

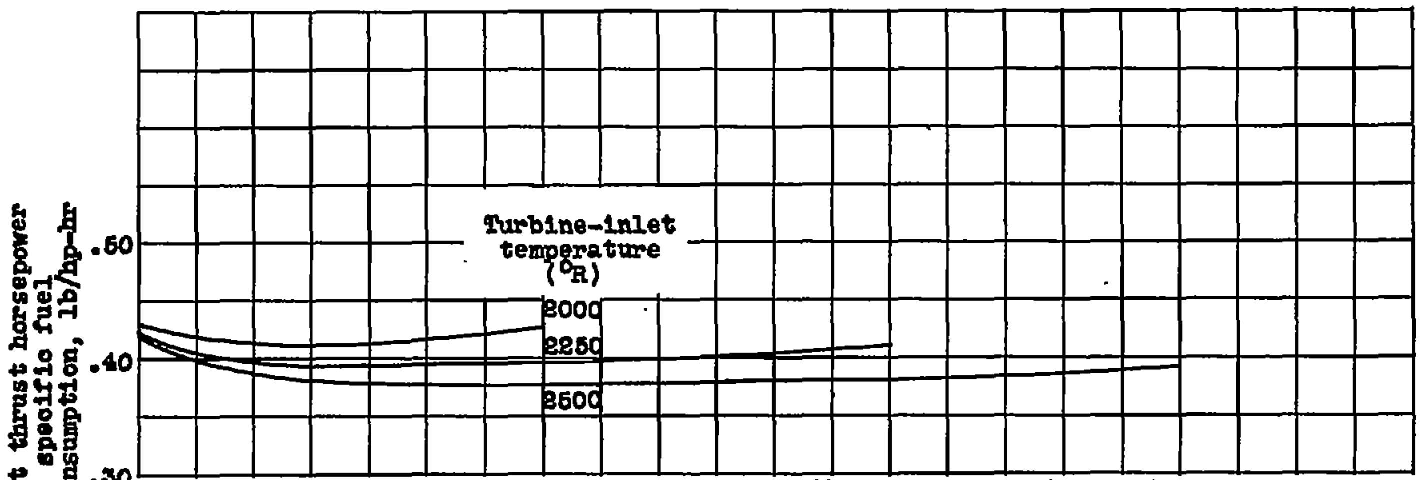


(a) Variation of net thrust horsepower specific fuel consumption with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R; 100-percent reheat; regenerative effectiveness, 0.5.

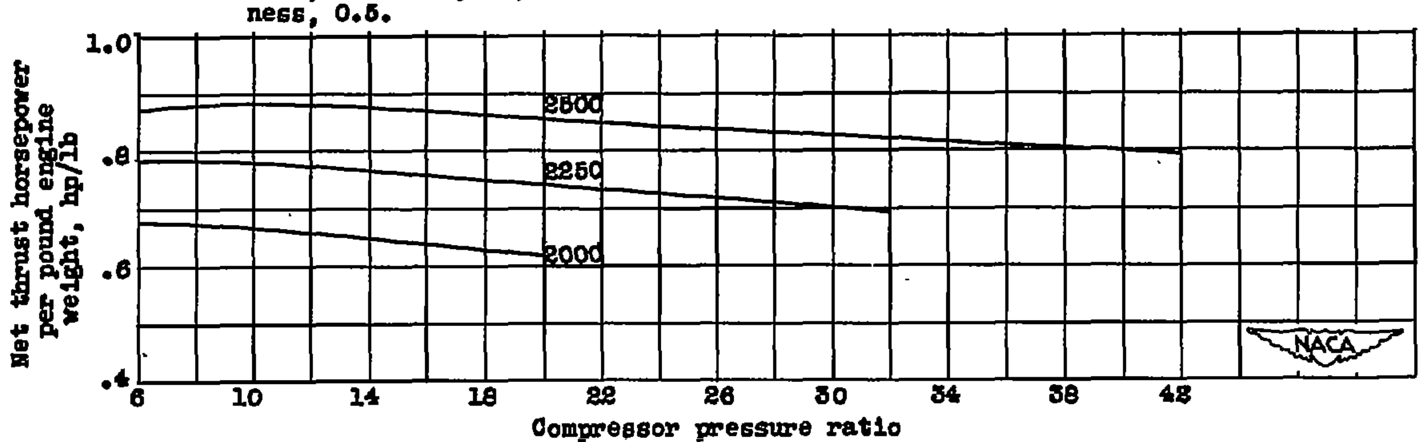


(b) Variation of net thrust horsepower per pound of engine weight with compressor pressure ratio, flight speed, and altitude. Turbine-inlet temperature, 2000° R; 100-percent reheat; regenerative effectiveness, 0.5.

Figure 5. - Performance of regenerative-plus-reheat turbine-propeller engine.

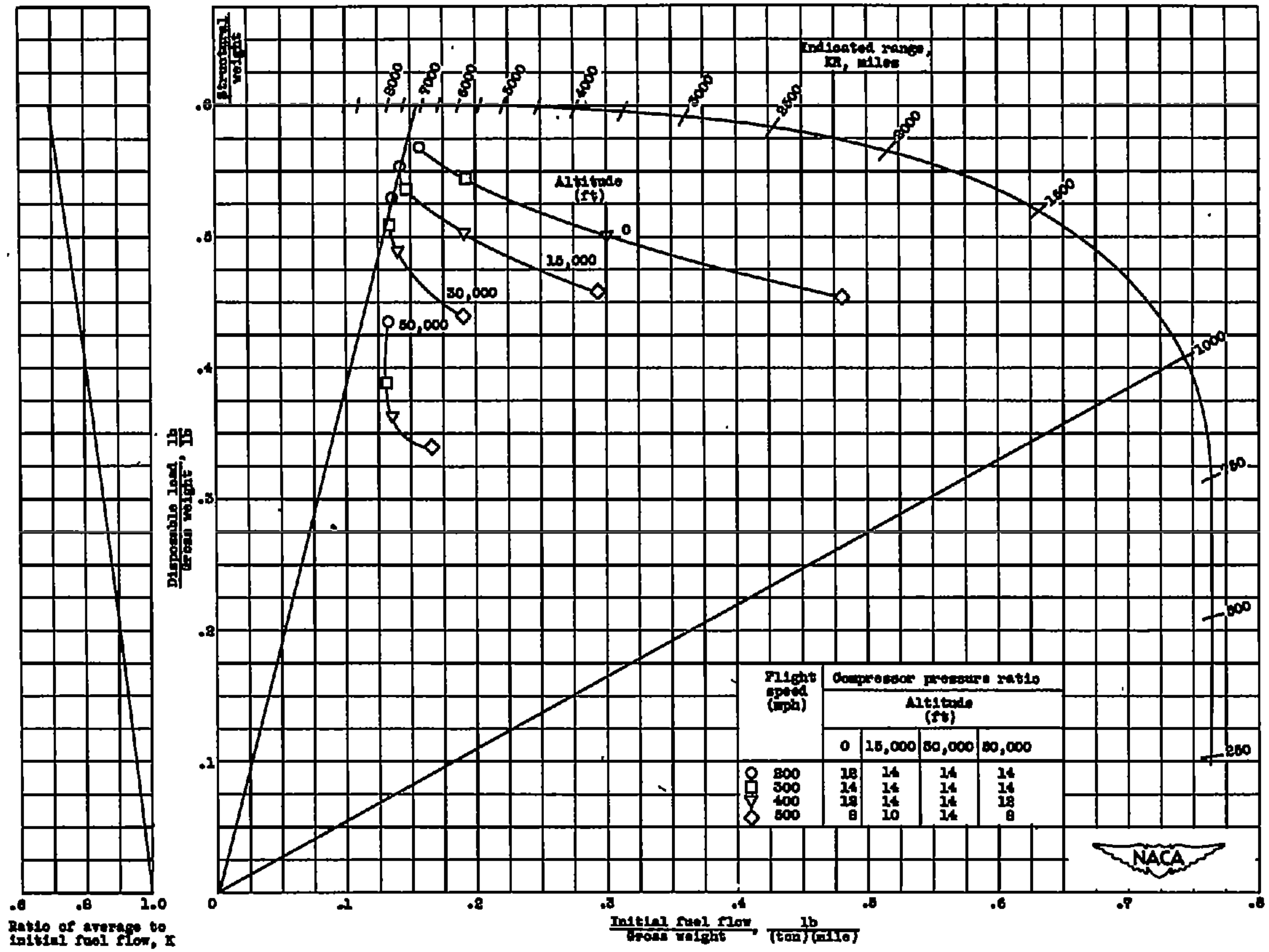


(c) Variation of net thrust horsepower specific fuel consumption with compressor pressure ratio and turbine-inlet temperature. Flight speed, 500 miles per hour; altitude, 30,000 feet; 100-percent reheat; regenerative effectiveness, 0.5.



(d) Variation of net thrust horsepower per pound of engine weight with compressor pressure ratio and turbine-inlet temperature. Flight speed, 500 miles per hour; altitude, 30,000 feet; 100-percent reheat; regenerative effectiveness, 0.5.

Figure 5. - Concluded. Performance of regenerative-plus-reheat turbine-propeller engine.



(a) Turbine-inlet temperature, 2000° R.

Figure 6. - Airplane range characteristics of basic turbine-propeller engine.

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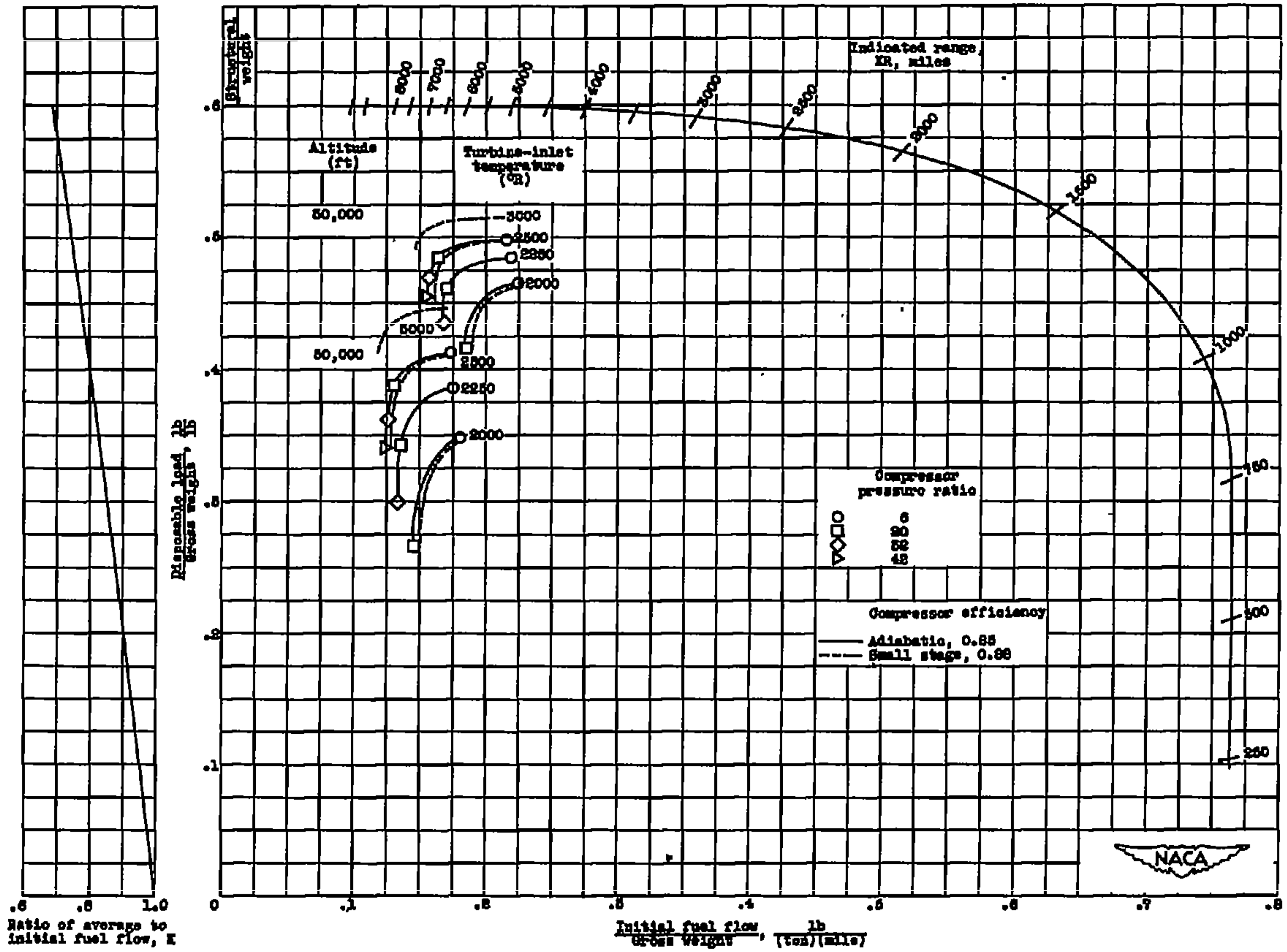
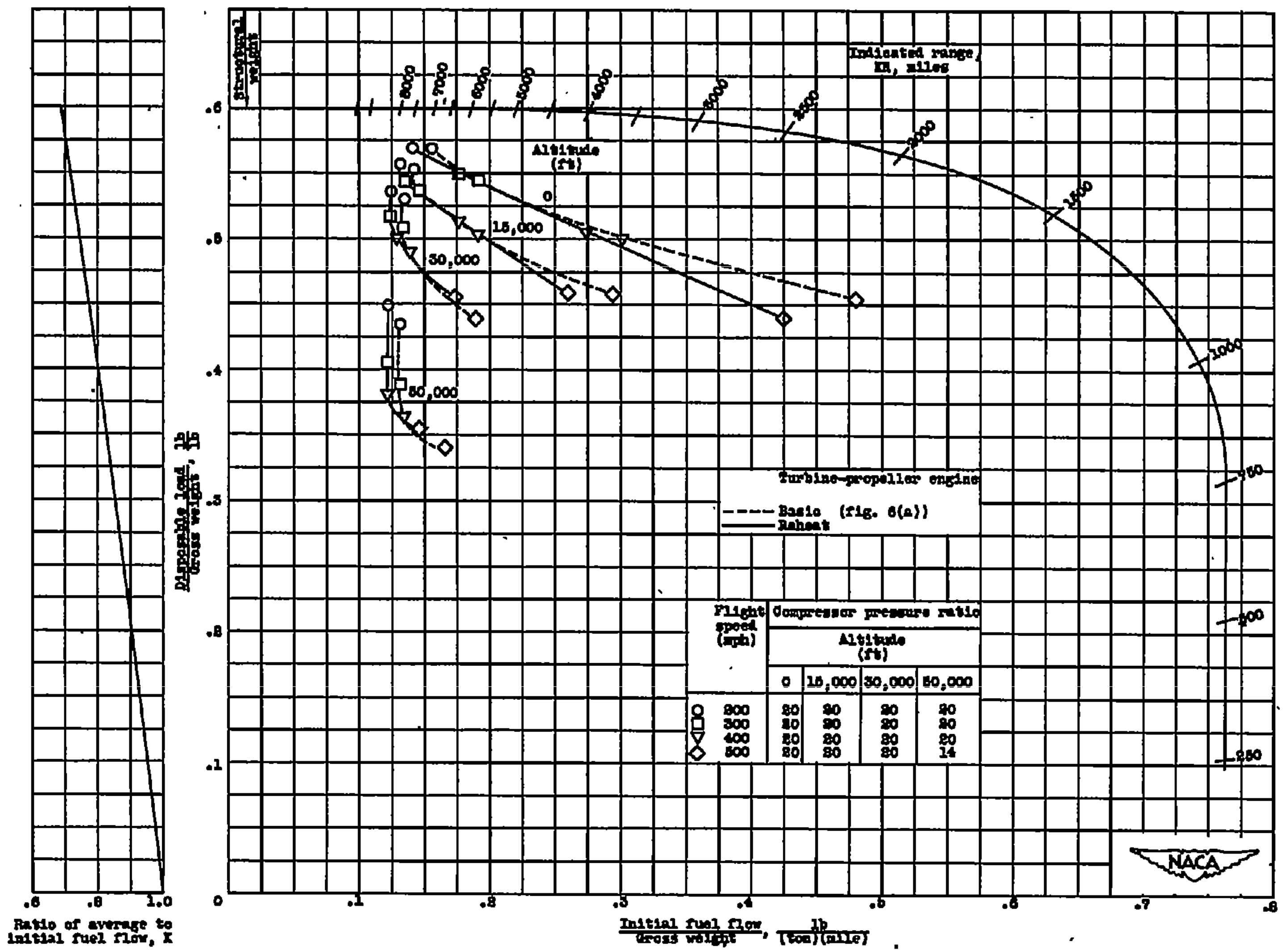


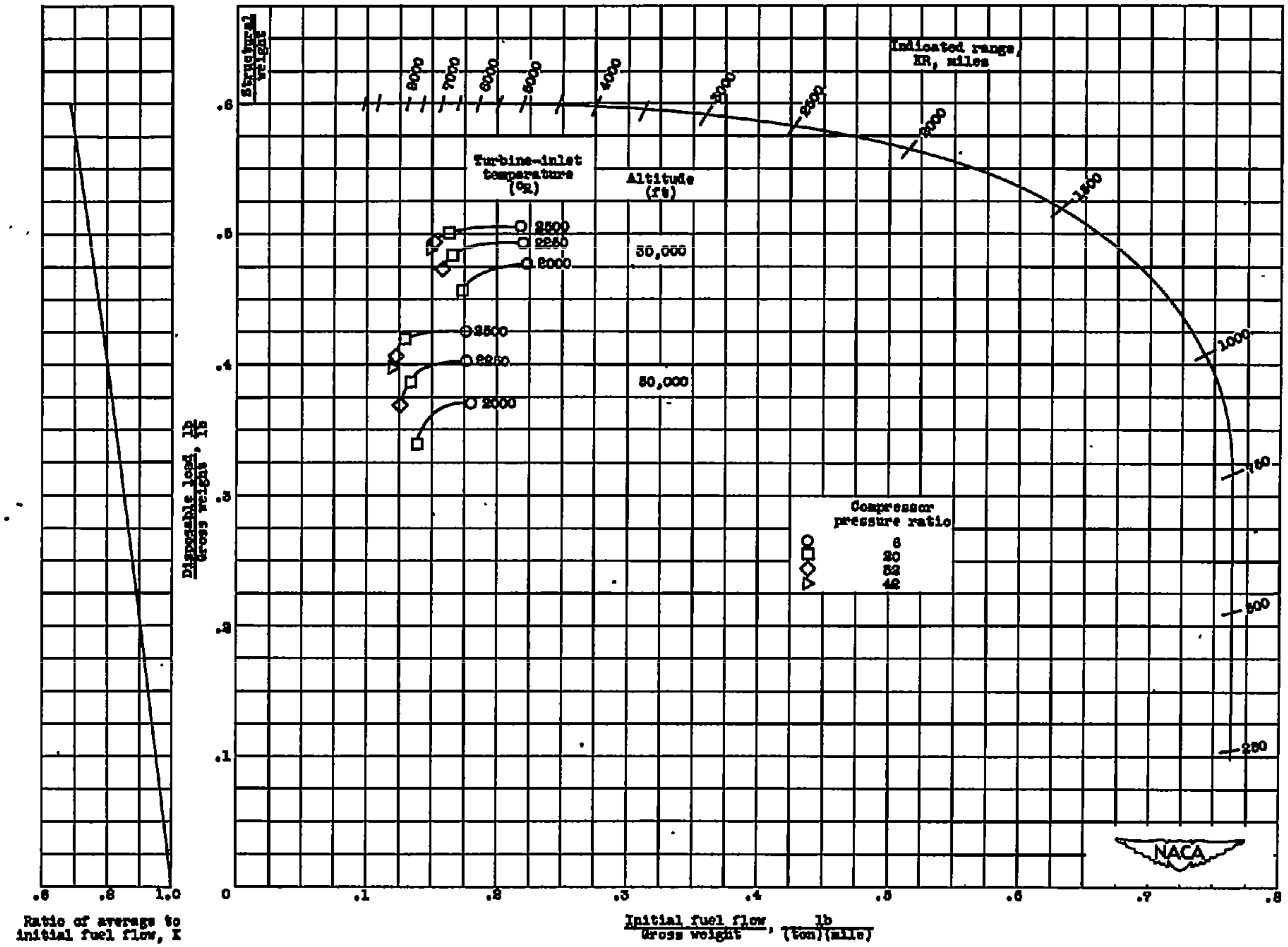
(b) Various turbine-inlet temperatures for cases of adiabatic and small-stage compressor efficiencies; flight speed, 500 miles per hour; altitude, 30,000 and 50,000 feet.

Figure 6. - Concluded. Airplane range characteristics of basic turbine-propeller engine.



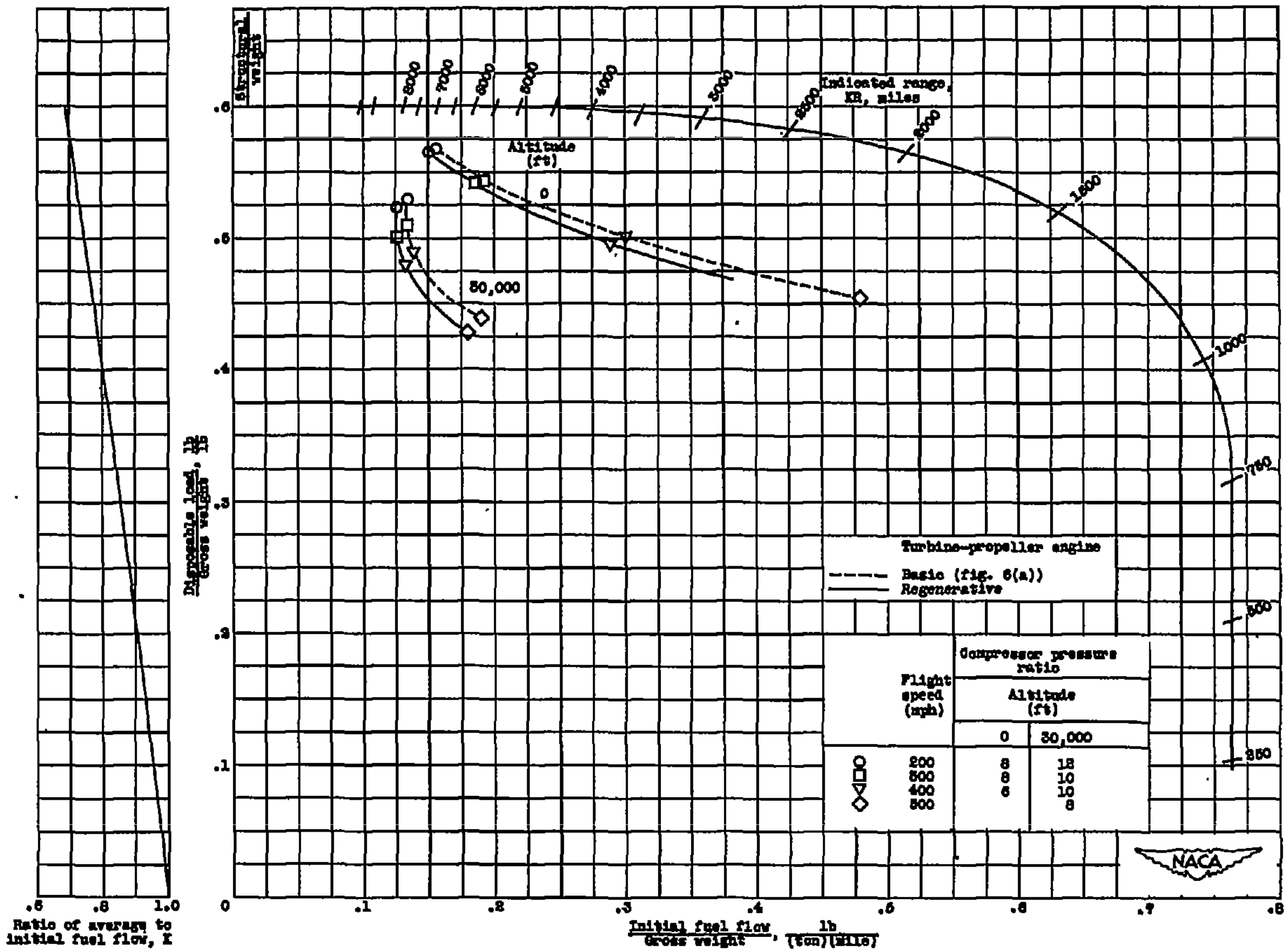
(a) Turbine-inlet temperature, 2000° R.

Figure 7. - Airplane range characteristics of reheat turbine-propeller engine.



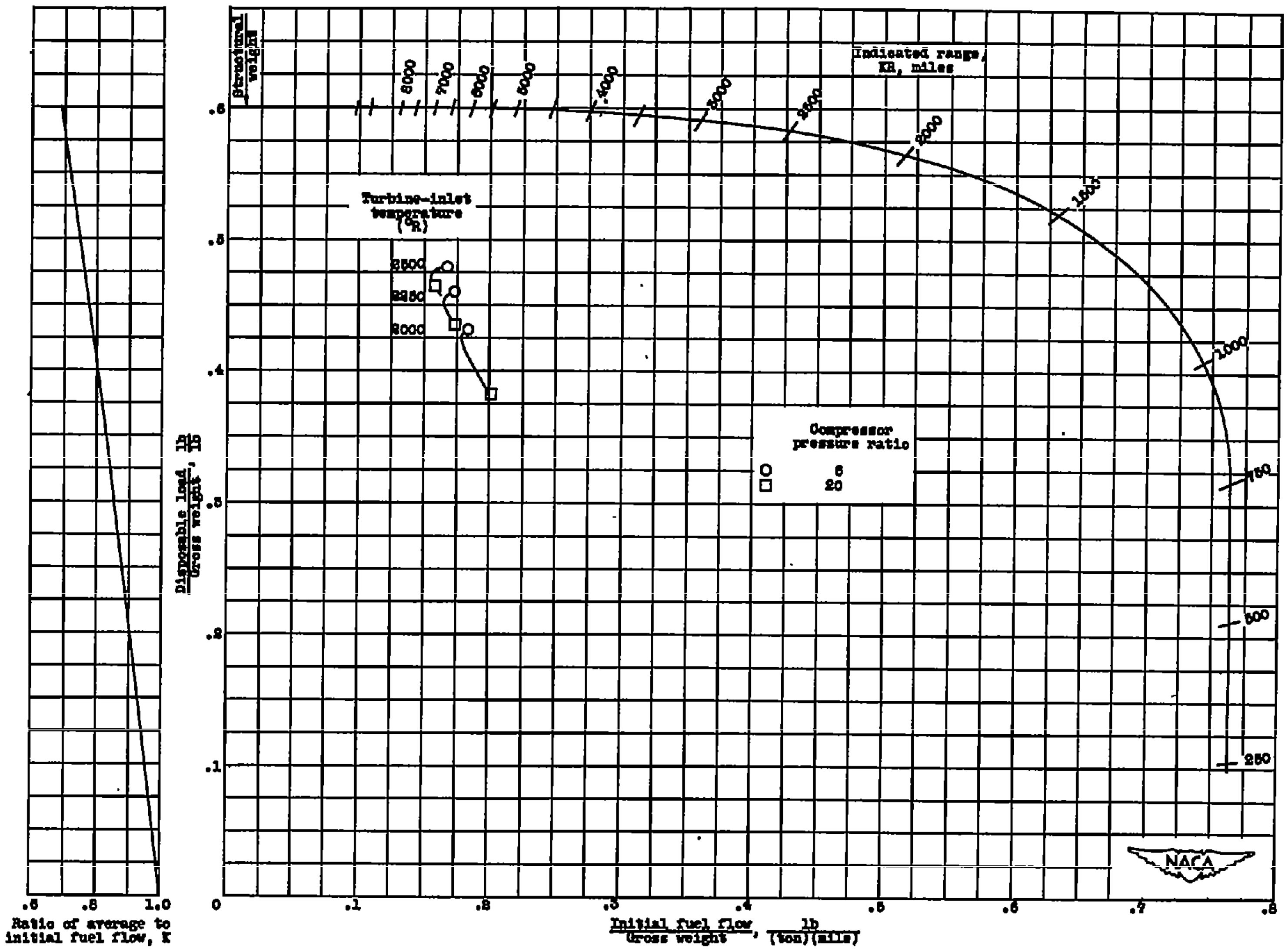
(b) Various turbine-inlet temperatures; flight speed, 500 miles per hour; altitude, 30,000 and 50,000 feet.

Figure 7. - Concluded. Airplane range characteristics of reheat turbine-propeller engine.



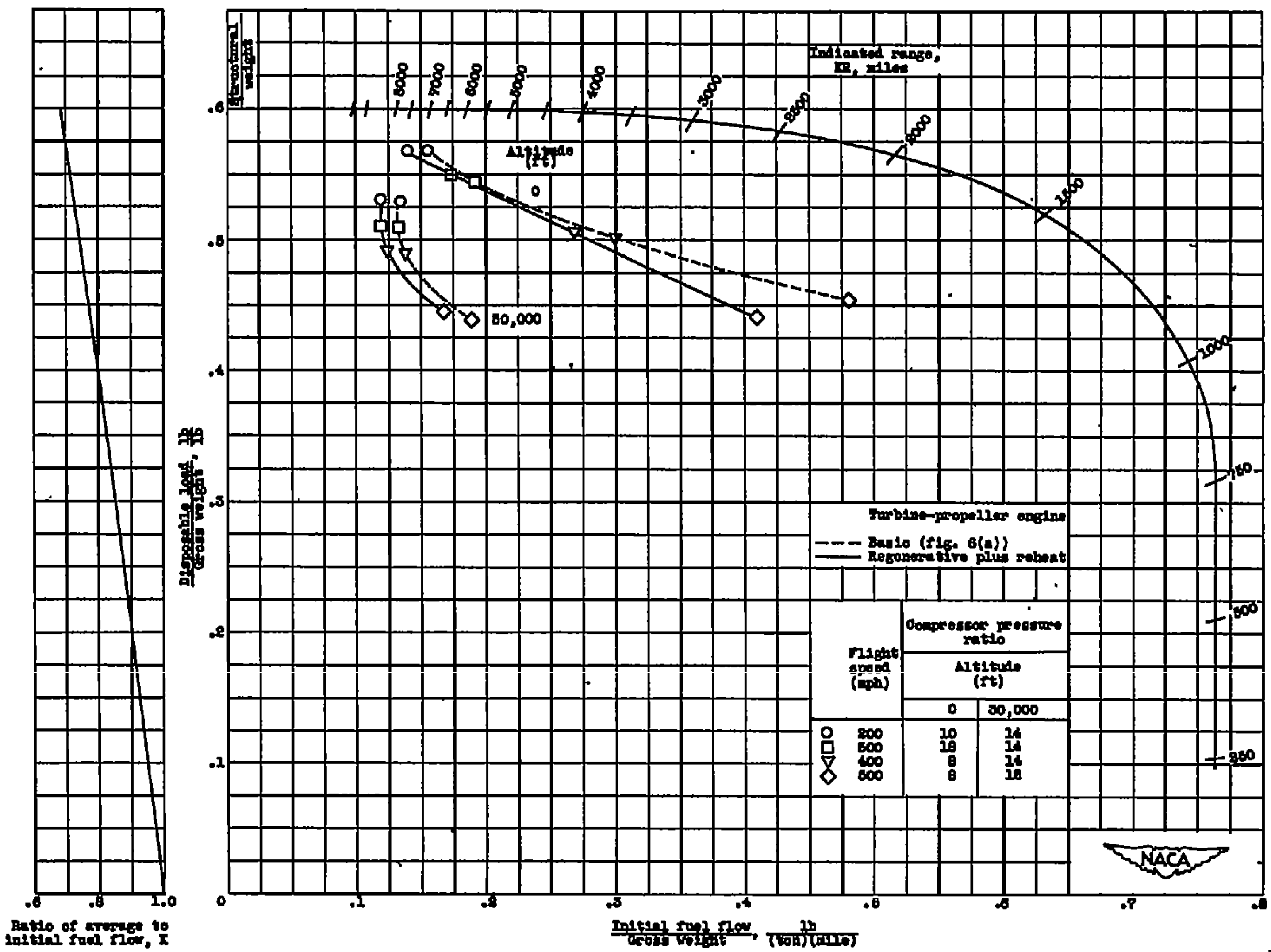
(a) Turbine-inlet temperature, 2000° R.

Figure 8. - Airplane range characteristics of regenerative turbine-propeller engine.



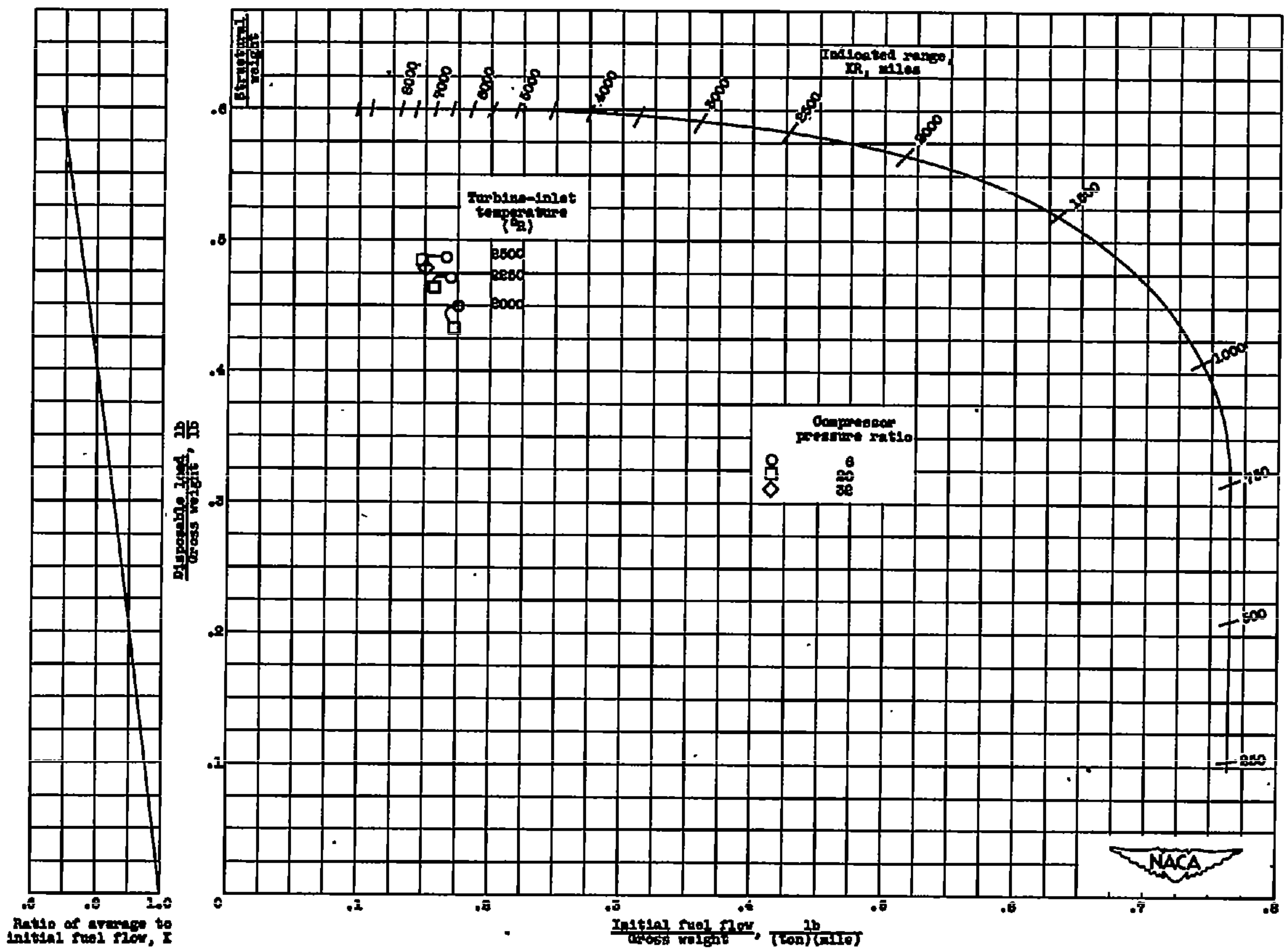
(b) Various turbine-inlet temperatures; flight speed, 500 miles per hour; altitude, 30,000 feet.

Figure 8. - Concluded. Airplane range characteristics of regenerative turbine-propeller engines.



(a) Turbine-inlet temperature, 2000° R.

Figure 9. - Airplane range characteristics of regenerative-plus-reheat turbine-propeller engine.



(b) Various turbine-inlet temperatures; flight speed, 500 miles per hour; altitude, 30,000 feet.

Figure 9. - Concluded. Airplane range characteristics of regenerative-plus-reheat turbine-propeller engine.